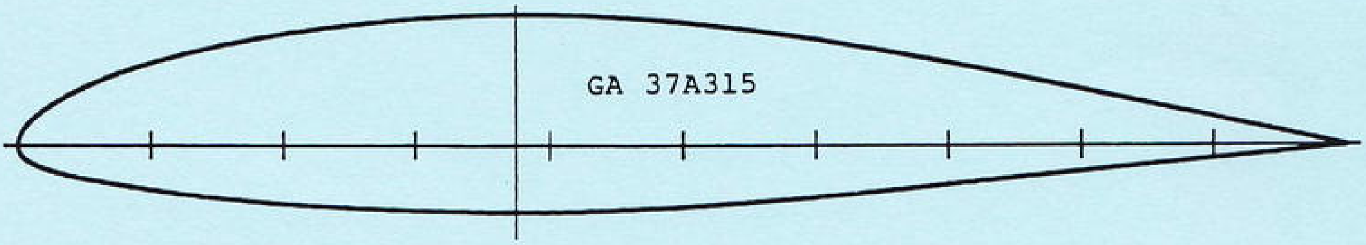
GA Airfoils

A CATALOG OF AIRFOILS FOR GENERAL AVIATION USE



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Preface to the Sixth Edition, February 1996.

The First Edition of "GA Airfoils", March 1987, contained only 30 airfoils. Subsequent editions added more airfoils to the matrix. This Sixth Edition expands the GA airfoil matrix yet again, with the addition of the high-cambered (-6xx) airfoils, and includes general revisions to the text. The discussion of trim drag, and the derivation of the trim drag formula, has been minimized, for this is primarily a concern of airfoil design rather than aircraft design. The airfoils of all previous editions are still good, valid airfoils.

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Introduction

"The NACA airfoils were not designed for actual airplane use, but were merely wind tunnel test specimens that were used to investigate the effects of changing various airfoil parameters on airfoil performance".

Unbelievable as this statement may appear to be, it is nonetheless true. I came to realize this around 1988, a year or so after beginning my serious investigation of the NACA sub-sonic airfoil work of the 1930's. In fact, about the time I began to realize that there were flaws and omissions in the NACA airfoil work, the statement above was quoted to me verbatim by a contractor-engineer who was then working for NACA Langley, and then things began to fall into place. I resolved to address the major mistakes and omissions in the work, and, while not straying too far from the NACA methods, "clean up" the NACA airfoils for the purpose of not only making them suitable, but outstanding for actual airplane use. The result is "GA Airfoils- a Catalog of Airfoils for General Aviation Use".

The "GA" airfoil matrix consists of: (example- GA 37A315)

- 4 series, GA 30, 35, 37, and 40. These numbers refer to the position of maximum airfoil thickness, in per cent of chord. They are derived from and replace the NACA 4-digit, 63 series, 64 series, and 65 series airfoils, respectively. The NACA 5-digit airfoils should not be used, for reasons discussed in the text below.
- 2 types, xx-xxx (cusp type), and xxAxxx (no cusp).
- 4 camber levels, -2xx, -3xx, -4xx, and -6xx, indicating the approximate design lift coefficient in tenths.
- 3 thicknesses, 12%, 15%, and 18% thick, as a percentage of the chord length. Other thicknesses within this range may be interpolated by scaling the thickness distribution up or down to the desired thickness, then adding the appropriate camber.

Thus the matrix contains 96 airfoils, plus 2 more (GA 30-613.5, and GA 30-616.5) shown on figure III-19. The matrix should cover the majority of GA needs for powered aircraft.

GA airfoils are not appropriate for helicopters or flying wing airplanes, both of which require special zero pitching moment airfoils. They are also probably not appropriate for sailplanes. GA airfoils are all forward loaded to produce low pitching moments, whereas sailplanes do better with mid-loaded airfoils. Mid-loaded airfoils produce the most lift with the least amount of camber, but they also produce relatively high pitching moments. Sailplanes are tolerant of high pitching moments, but typical powered airplanes are not.

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1. SUMMARY

A catalog of 98 newly-designed "GA" subsonic airfoils along with computer derived performance prediction data is presented herein for General Aviation applications, that is, for flow conditions generally in the range of Reynolds numbers from one to ten million, and for aircraft speeds to 300 MPH. These airfoils correct the errors, shortcomings, and omissions contained in the NACA airfoils presented in reference 1, "Theory of Wing Sections", and later NASA subsonic airfoils, and are designed to replace those airfoils completely. Compared to those government "airfoils", this author's "GA" airfoils offer softer stalls and better slow-speed performance, better aileron and flap effectiveness, and lower pitching moment coefficients, with no decrease in high speed performance. Contrary to the NACA/NASA airfoils, "GA" airfoils are presented in a logical, integrated matrix spanning a wide range of turbulent and laminar flow type designs. Predicted performance of all sections has been verified using reference 6 computer program, and the results are presented and compared with typical corresponding NACA/NASA airfoils. The GA airfoils are evolutionary rather than revolutionary, and are conservatively designed. They are the next step which NACA should have, but did not take, following their wind tunnel tests to determine the effects of changing certain airfoil parameters, summarized in reference 1. GA airfoils are designed for use on real airplanes, not merely as wind tunnel test specimens.

We will now summarize the NACA airfoil test work from a historical perspective, and then rate the features tested, and discuss certain features that were not tested, so that we may combine the best features into our GA airfoils.

2. HISTORIC NACA AIRFOIL TEST WORK

The National Advisory Committee for Aeronautics was established by Congress in the early 1920's, shortly after the end of WWI, to indirectly subsidize the infant U.S. aviation industry. WWI had shown that the U.S. lagged behind other modern countries in aviation development; in fact, the only significant aeronautical contributions by the U.S. in WWI had been the Curtis JN-1 trainer, and the Liberty engine, while other countries had perfected a wide range of high performance airplanes and engines. For this reason, Congress decided to fund a program to conduct basic aeronautical research, with the hopes that the infant U.S. aviation industry could use this knowledge to "catch up". Central to this NACA research effort was the construction of modern wind tunnel test facilities at Langley, Virginia, the Langley Memorial Aeronautical Laboratory (LMAL). By the mid-1920's, this laboratory was on-line, and NACA began testing known airfoils from all over the world, not only to calibrate the tunnels, but to conduct a technology search. This was by nature random testing. The next step was to develop a systematic airfoil test program, varying certain airfoil parameters to determine the effect on airfoil performance. The best known of these test airfoils were the 4 digit, 5 digit, and 6-series airfoils. This arduous test work continued through WWII, and the amount of test data that was collected is impressive. Unfortunately, however, since NACA never did follow up this test work with catalogs of airfoils specifically designed for aircraft use, the test airfoils came to be used on a wide range of airplanes, with mixed results.

In more recent years, NASA has published certain isolated airfoils from time to time, usually a personal effort by individual NASA employees. These have all had poor results, and all of them require extensive modification to make usable airfoils from them. Examples include the NASA GAW-1 and GAW-2 [LS(1)-0417 and LS(1)-0413), NLF(1)-0215F, and NLF(1)-0414F].

A. NACA 4 digit airfoils. For these systematic tests, a series of test airfoils was designed by the classic method of determining a symmetrical thickness distribution, and then curving this "basic thickness form" with a carefully designed camber schedule known as the "mean line", to obtain a cambered airfoil. Reference 1 is a summary of the NACA airfoil test work and follows this pattern: Appendix I contains the basic thickness forms, Appendix II contains the mean lines (camber profiles), Appendix III contains the ordinate tables for the cambered airfoils, and Appendix IV presents the wind tunnel test results. "GA Airfoils" follows this same general pattern.

A designation system was then devised, wherein the first digit indicates the maximum camber in % of chord, the second digit indicates the location of maximum camber in tenths of the chord length, and the last two digits indicate the maximum thickness of the airfoil as a percentage of the chord length. For example, NACA 2412 indicates a 12% thick airfoil with 2% maximum camber located at 40% of chord. Note that the maximum thickness of this airfoil, as in all of the 4 and 5 digit airfoils (turbulent), is at 30% of chord, as determined by the general thickness distribution formula given below. With this system in place, an orderly family of airfoils was constructed and tested, with various thicknesses, camber levels, and chordwise position of maximum camber. Other variables that were tested, not included in the designation system, included conditions of "standard roughness", simulated split flaps deflected 60 degrees, and Reynolds numbers of 3, 6, and 9 million.

The first step was to design a carefully specified thickness distribution. The NACA researchers noted from earlier tests that the best airfoils, such as the German Gottingen #398 and the U.S. Clark Y, had the maximum thickness at 30% of chord. Accordingly, the thickness distribution for the 4 digit series was selected to correspond closely to that for those wing sections, and is given by the following equation:

$$\pm Y = \frac{t}{.20}\left(.2969\, X^{\frac{1}{2}} - .126\, X - .351\, X^{2} + .2843\, X^{3} - .1015\, X^{4}\right)$$

where t is the maximum thickness expressed as a fraction of the chord. In practice, the ordinate schedule for the 20% thick section (0020) is determined from this formula, and then the ordinate schedule for the other thicknesses is determined by scaling the 0020 ordinates directly up or down to the desired thickness.

For the camber schedule, a family of parabolic curves (mean lines) was developed, designated as mean lines 62 through 67, indicating that the maximum camber of 6% of chord was located at the 20, 30, 40,50, 60, and 70 per cent chord position. This gave a selection of forward loaded (62-64), mid-loaded (65), and aft loaded (66&67) mean lines. For less camber, the basic 6% mean lines are simply scaled down directly to the new value.

Now the basic thickness forms and the camber schedules were combined according to the "NACA" method, shown in figure III-1 below, which turns out to be a faulty method, and this was one of the major mistakes of the NACA researchers. More on that later. In any event, the 4 digit "airfoils" so obtained were tested, and since the advantages of forward loading had been demonstrated previously, the most practical of these airfoils used the 64 mean line, which placed the maximum camber at 40% of chord (NACA x4xx). These airfoils, in spite of their shortcomings, were widely used on popular airplanes of the day, such as Cessna (2412), Luscombe and Aeronca (4412), and Seabee (4415), etc.

B. NACA 5-digit Airfoils. Following the 4-digit airfoil work, NASA conducted a test program in the early 1930's to see if zero pitching moments could be achieved on cambered airfoils by using leading edge droop alone, rather than using a conventional camber schedule. (Read Addendum #1 below, "Understanding Airfoils", for a discussion of the relationship between Cl, Cd, and Cm in cambered and uncambered airfoils). Zero pitching moments on cambered airfoils had been achieved earlier by reflexing the airfoil trailing edge, as on the M-6 and $2R_2 12$ airfoils, but this method has the disadvantage of reducing the maximum lift coefficient. The new test airfoils, using leading edge droop alone, became known as the "5-digit" series. In doing these tests, NACA shared the general fascination of that era for zero pitching moment airfoils, without critically evaluating the need for this feature, and without quantifying the effects of differences in airfoil Cm on aircraft performance.

The only airfoils of any consequence in this series are the 230xx airfoils, represented by the ubiquitous 23012 airfoil, which is 12% thick, and has 1.8%C leading edge droop forward of the 15% chord position. The airfoil has no other camber, and the mean line is perfectly straight from .15C to the trailing edge. The thickness distribution for all of the 5-digit airfoils (0012 etc) is the same as for all of the 4-digit airfoils. The same faulty NACA method of combining thickness and camber (droop) ordinates was also used, and due to this and the "slope and radius" method for laying out the leading edge radius, the effective "camber" for NACA 23012 is only 1.4% rather than 1.8%C. Thus, the airfoil is especially low cambered, and this combined with the extreme forward loading from leading edge droop alone accounts for its very low Cm, meeting the design objective. The airfoil even has a maximum lift coefficient equal to or greater than comparable 4-digit airfoils, due to the "separation bubble" lift phenomenon discussed in Addendum #3 below. However, the airfoil has a major disadvantage- an unacceptable sharp stall,caused by the discontinuity (kink) in the mean line at .15C. The NACA 5-digit airfoil experiments proved that zero pitching moments as well as reasonably high lift coefficients could be achieved by leading edge droop alone, but the price paid (sharp stalls) is completely unacceptable for general aviation applications. Accordingly, the NACA 5-digit airfoils should be regarded as nothing more than laboratory curiosities, not to be used on actual aircraft.

It is unfortunate that the NACA test data on these airfoils did not include the hysteresis loop that occurs with decreasing angle of attack after the sharp stall has occured, as in attemted recovery from the stall. In fact, none of the NACA airfoil test work included such post-stall decreasing alpha tests, which would have illustrated the insidious nature of sharp-stall airfoils, and could have saved many lives, especially on twin-engine airplanes. After a sharp stall has occured, a considerable reduction of angle of attack is necessary (with consequent loss of altitude) to re-establish attached flow on the airfoil top surface, meanwhile the aircraft is completely without lateral control. The failure of NASA to eliminate sharp-stall airfoils from their repertoire, or at least to issue strong warnings against their use, borders on criminal neglect.

Another reason beside the safety concern for abandoning the 5-digit airfoils is that there is a better way to get higher performance airfoils, that is, by using laminar flow techniques, discussed below.

C. NACA 6-Series Airfoils. The next significant NACA development was the laminar flow work of the late 1930's and 1940's, resulting in the 6-series, or laminar flow, airfoils.

The two major flow conditions in the field of fluid mechanics are laminar flow, and turbulet flow, which has approximately twice as much drag as laminar flow.

To remain laminar, the flow must be unaccelerated, vertically as well as chordwise (constant velocity), within rather narrow limits. Therefore, NACA researchers devised special "constant velocity" shapes, characterized by relatively slender nose sections and smaller leading edge radii, and having the position of maximum thickness moved aft from 30% of chord to 35%, 37.5%, 40%, and 45% of chord for the 63-, 64-, 65-, and 66-series shapes, respectively. These shapes exhibit the ability to maintain long laminar runs through an angle of attack range of approximately plus or minus 3 degrees from zero before the flow trips to turbulent. This low-drag range is known as the "laminar bucket". These shapes also use "cusps" in the critically shorter afterbody, or "pressure recovery" section, to improve efficiency in that area. NACA also devised a series of constant velocity mean lines to go along with the constant velocity thickness distributions, to replace the parabolic (4-digit) mean lines. This work is discussed in Abbott and Von Doenhoff's "Theory of Wing Sections", reference 1, which is a must for all serious students of aeronautics.

With these 6-series airfoils, laminar flow can be achieved rather easily on the top surface back to and slightly beyond the thickest part of the wing, before the flow "trips" to turbulent flow at the "transition point". In fact, NASA reports in reference 8 that the extent of laminar flow with these airfoils, even on production airplanes, is greater than had been believed previously, and is quite stable. Thus, drag reduction for the wings on the order of 25% compared to a similar turbulent wing is not unreasonable. For example, the section drag reduction for the NACA 64-212 airfoil is 25% less than NACA 23012. This is a significant drag reduction, and for these reasons laminar flow airfoils should have revolutionized the general aviation industry, especially considering the fact that these airplanes are in a Reynolds number range that is especially favorable for laminar flow airfoils. Furthermore, the constant velocity mean lines developed by NACA for these airfoils includes one (a=0.5 mean line) that produces lower pitching moments through forward loading, with no drag penalty, a well-proven technique that was used in the prior (turbulent) airfoils. This mean line is designed to hold constant pressure (therefore constant velocity) back to only 50% of chord, which is the practical limit for laminar flow anyway, and then it tapers the loading linearly to zero at the trailing edge. For anyone acquainted with the NACA airfoil design methods, and willing to derive their own airfoils, this seems to offer the "best of both worlds", that is, low Cm as well as low drag compared to the published "base line" 6-series airfoils, which have relatively high Cm due to mid-loading. In fact, Cessna did just that, by deriving and using the unpublished NACA 64A215(a=0.5) for the root, and the unpublished NACA 64A412(a=0.5) for the tip, of the Model 210 "Centurion" wing.

In spite of the potential performance improvement that could be made by using laminar flow wings, they have never gained wide acceptance in the GA field. Most small light planes continue to use the 4-digit and 5-digit turbulent airfoils, and the great majority of light twins, and even commuter transports, use the obsolete and dangerous 5-digit airfoils, a horrible mis-match considering the consequences of sharp-stall airfoils in an engine-out situation. The NACA 6-series airfoils are often avoided, since they have gained a reputation of being "killer airfoils" with sharp stall characteristics and low maximum lift coefficients, as well as relatively high pitching moment coefficients. Read on, and you will see that this unfortunate situation has nothing at all to do with the fact that these airfoils are laminar flow-type airfoils. NACA simply screwed up.

3. GA AIRFOIL DEVELOPMENT

A. Ordinate Combination Method. All of the NACA airfoils, summarized on figure 1, were designed by the classic method of combining a thickness distribution with a camber schedule to form a cambered airfoil. An obvious mistake in the NACA airfoil development work was the method used for combining the thickness distribution ordinates with the camber ordinates. There are two ways to do this, the right way and the wrong way. Von Mises discusses both methods in reference 2, but unfortunately he does not indicate a clear preference for one or the other. The first (correct) method is simply to add the ordinates together at each wing station. The second (incorrect) method, which for some unknown reason appealed to NACA, is akin to rolling a circle of ever-changing radius along a curved mean line, top and bottom, and then the area swept by the rolling circle becomes the airfoil shape. This method, shown on figure III-1 below, is quite complicated mathematically, which ironically may be the reason that NACA picked it. Unfortunately, it leads to a distortion at the leading edge, which is then smoothed by the so-called "slope and radius" method for forming the leading edge. This faulty process is used on all NACA airfoils. Unfortunately, this has the bad effect of super-elevating the leading edge above the original chord line. Figure III-1 shows an example- with the NACA 4415 airfoil the leading edge is lifted .486C above the original chord line, creating a new mean line above the original mean line. This de-cambers the airfoil, as a function of the airfoil thickness, as shown by figure 5. As a result, the thinner NACA airfoils have more camber, and thus more lift, than the thicker ones. This may explain NACA's fascination with thinner airfoils, especially for wing tips- another mistake. See Addendum Number 2 below. Worse yet, the incorrect method of combining ordinates flattens the initial slope of the mean line, and this detracts from the low-speed performance of the airfoil. But NACA never investigated this effect.

Figure 2 shows the beneficial effect of using the correct method of ordinate combination on a typical NACA airfoil, the ubiquitous NACA 64-212. When the correct method is used to form the airfoil, redesignated 64-212C, the stall is softened appreciably, and the maximum lift coefficient with and without flaps is raised slightly, compared to the original airfoil. Accordingly, the "direct addition" method of ordinate combination is used on all "GA" airfoils.

B. Mean Lines. Figure 3 shows the wide variety of mean lines that have been used in NACA/NASA airfoils, from extremely forward-loaded ones (230xx) to aft-loaded ones (GAW-1,-2). Most of the popular NACA 4-digit airfoils use moderately forward-loaded mean lines (x4xx-see figure 1) with maximum camber at 40% of chord, since it was known prior to 1930 that forward loading produces low pitching moment coefficients, and this had been established in general terms as a desirable airfoil characteristic. In spite of this, only 5 of the 62 6-series airfoils listed on figure 1 have forward loaded mean lines (a=0.5,0.6), while all the rest have the exactly mid-loaded (a=1) mean line. Why the contradiction? Simply because these airfoils are merely test airfoils, not designed for actual airplane use. Only enough forward-loaded airfoils (five) were included in the test program to confirm the effects of that variable. Further, the relative importance of the airfoil pitching moment coefficient (Cm) compared to the drag coefficient (Cd) has never been studied nor quantified by NACA or NASA. How can one intelligently design airfoils without knowing this relationship? For example, figure 6 shows three airfoils with various combinations of drag coefficient and moment coefficient. How important is it to have low Cm, relative to Cd? Is it all-important, or just a nicety? Do we need zero Cm?

To answer these questions, we investigated the drag that is caused by airfoil Cm, and derived a formula (figure 8) for calculating this "trim drag", or drag associated with trimming out the airfoil pitching moment, for a typically configured light plane. When trim drag is added to the airfoil section drag, we arrive at a "corrected" drag coefficient, Cd', which can now be used directly for comparing candidate airfoils for a given application.

Briefly, the problem arises due to the fact that the center of pressure of cambered airfoils moves aft from the C/4 position as the angle of attack decreases- see figure 7. This causes an aircraft nose-down pitching moment at higher speeds that must be reacted by negative lift in the horizontal tail. Remember that the aircraft CG must stay at or near the C/4 position for satisfactory low-speed pitch stability. But the negative lift in the tail causes induced drag. Also, the wing must now generate an additional increment of lift equal to the tail down-load, since the summation of vertical forces on the airplane must be zero. This additional increment of wing lift also creates an increment of induced drag. Add these two additional drag increments (wing and tail) together, and we have trim drag. Notice from figure 8 that the trim drag coefficient for the very high-Cm NASA GAW airfoils is approximately 20 drag counts at cruise for an application having a design lift coefficient of 0.2. When this is added to the section drag coefficient of 53 drag counts, we obtain a corrected drag coefficient of 73 drag counts. This renders the NASA GAW airfoils unsatisfactory, or at least very disappointing, for this type of application.

As a result of analyses like this, we have concluded that forward loading is a necessity for general aviation airfoils. However, it is not necessary to go to the ridiculous extreme of the ill-advised 5-digit airfoil test program ("airfoils having the maximum camber unusually far forward"). In short, it is not necessary to have zero Cm, nor is it necessary to reduce Cm to the point where other vital performance characteristics of the airfoil are compromised. The NACA 5-digit airfoils achieved zero Cm, but the price paid (sharp stall) is unacceptable, therefore the 5-digit airfoils should not be used.

Therefore, we have selected the NACA (a=0.5) mean line to be the basis for the universal family of only four camber profiles for all of the "GA" airfoils, turbulent as well as laminar flow type. The a=.5 mean line, when substituted in the NACA 64-212C airfoil, produces 64-212(a=.5)C. Figure 2 shows that this results in a wider laminar bucket, slightly higher maximum lift coefficient with flaps, and most importantly, a 20% reduction of pitching moment coefficient, with no other penalty.

The next step is to control the initial slope of the mean line, an important parameter for good slow-speed performance. This was completely missed in the NACA work, although some later NASA airfoils have leading edge droop, which corrects this oversight. We have found that the initial slope should be a minimum of 12 degrees for any airfoil, thus our GA-2 mean line incorporates .3% droop, and the GA-3 mean line has .2% leading edge droop- see Appendix II. The effectiveness of the droop can be seen in the performance of airfoil GA 37-212, which has a much softer stall, an even wider laminar bucket, low Cm, and yet higher Clmax with flaps, with no drag penalty, as shown on figure 2. For this reason, all GA airfoils can be described as "low pitching moment, soft-stall airfoils". In fact, merely by selecting a GA airfoil, an aircraft designer can be assured that the pitching moment coefficient has been held as low as practical, short of degrading other performance features of the airfoil, thus for the majority of cases a trim drag analysis is not required. However, in extreme cases, such as a very high performance application with the design lift coefficient less than .15, a formal trim drag analysis may be advisable.

Forward loading has one possible disadvantage, described on figure 10. Since forward loading increases the velocity ratio slightly in the forward half of the airfoil compared to mid-loading, the critical Mach number for these airfoils is degraded slightly. However, this occurs in the 580 MPH range, far above our needs, hence this is of no concern to us.

C. Thickness Distributions. For our GA airfoils, only minor changes have been made to the NACA thickness distributions. The GA thickness distributions are listed in Appendix I below. For the turbulent airfoils (GA 30), the GA 30A015 shape is identical to NACA 0015. However, this shape has a slightly convex, or "boat-tailed" afterbody, which is rather unusual. Therefore, we have generated an alternate turbulent shape, GA 30-015, with a straight afterbody, that reduces the profile drag from 75 drag counts to 72 drag counts, with no structural disadvantage. One concern here, however, is the change in the trailing edge angle, which affects aileron effectiveness-see figure 9. Since the trailing edge angle is reduced in GA 30-015, the control effectiveness may be slightly less than with GA 30A015. However, the trailing edge angle remains generous. Further, there are better ways to get effective ailerons, such as making "fat" or even "crowned" ailerons, which can then be used with the GA 30-xxx airfoils.

For the GA 35 and GA 37 laminar shapes, no changes were made to the NACA 63 and 64 series shapes, which are conservative, utilitarian laminar sections, having wide laminar buckets. This allows airplanes with these thickness distributions to operate over a wide range of flight conditions, while still remaining "in the bucket". This feature results in a profile drag penalty of perhaps only one or two drag counts compared to some other so-called "superior, low-drag" laminar thickness distributions that have been developed since, and the drag penalty is a small price to pay for the wider bucket on the classic NACA shapes.

For the GA 40-015 shape, we use the NACA 65-015 thickness distribution as is. The profile drag coefficient of this shape is also slightly higher than recent NASA "slab" shapes, but the drag bucket is correspondingly wider, for operational flexibility.

For the GA 40A015 shape, we have carefully avoided the NACA 65A015 shape, which has a poorly designed, straight afterbody (high drag) and should not be used. With the point of maximum thickness moved aft to the 40% chord position, the design of the pressure recovery section becomes critical, and a straight afterbody is no longer suitable, due to the unaccaptable drag penalty. Accordingly, we settled on a reduced cusp shape by modifying the NACA 65-015 to fill in approximately one-half of the cusp. This gives the desired structural and control effectiveness advantages of a true no-cusp shape, with a drag penalty of less than 2 drag counts.

Notice that the basic thickness distribution for the GA thickness distributions is 15% thick, and then the 12% and 18% thicknesses are scaled proportionately up or down from the 15% thick shape. This is a departure from the NACA scheme for their laminar thickness distributions, wherein unique symmetrical shapes have been calculated and published for each thickness, and these shapes are not quite exact multiples of each other. The difference is very small, however, and does not affect airfoil performance, therefore we have ignored it to simplify the process of generating families of airfoils.

Notice that the minimum section thickness offered in "GA Airfoils" is 12%, for we do not recommend the use of wing sections thinner than this for any application, even for racing airplanes. In fact, the optimum airfoil thickness regarding lift to drag ratio is probably closer to 13% or 14% thick, for any airfoil system. Simply put, you lose more than you gain by using wing sections thinner than 12% thick.

For tail sections, symmetrical shapes such as NACA 0009 or NACA 63A009 are commonly used. Ordinates for these shapes are listed in standard NACA tables (reference 1), or they may be scaled down from the corresponding GA thickness distribution.

D. Ordinate Tables. Notice from figure 1 that the NACA "ordinate tables" are merely a listing of test samples that were included in NACA's wind tunnel tests, and are thus incomplete. In contrast, the GA airfoil ordinate tables, Appendix III, are a complete matrix of 96 airfoils, in four series, two styles (with and without cusps), four camber levels, and three thicknesses, offering the designer a complete "cook book" of airfoils from which to choose.

NASA's official position is that a comprehensive catalog or "cookbook" of airfoils is not necessary, and suggests that a unique airfoil should be custom-designed for each and every new airplane, one that "exactly fits the specification for the new airplane", otherwise the performance of the new airplane will be compromised. This is, of course, nonsense. There is no single, unique solution to a particular airfoil design problem, since there are so many value judgements, intangibles, and performance trade-offs involved. For example, how "soft" is "soft", for a soft-stall airfoil? Therefore, there will always be a need for a good catalog of airfoils, if for no other reason than to serve as a standard to compare custom-designed airfoils against. Further, airplane designers are by-and-large not airfoil designers, and would rather select from a good catalog of standardized designs rather than take a chance on a custom airfoil.

NACA's thinking, evident from reading reference 1, was that "custom" NACA airfoils could be synthesized using NACA's data and methods. In other words, NACA assumed that a designer could select a particular thickness distribution and a camber profile from the text, and then combine them by the NACA method, to obtain the desired custom airfoil, just as Cessna did for the Model 210 wing design. This is probably the reason that NACA was content to publish test data alone, rather than to publish actual catalogs of airplane airfoils. However, to thus synthesize custom NACA airfoils is a daunting task, beyond the capability (or patience) of the average designer. Further, due to the mistakes and omissions in the NACA work, the airfoils would still be less than ideal. For example, Cessna's airfoils for the Model 210, while a step in the right direction, do not contain leading edge droop, and the ordinates are combined incorrectly, which is both ironic and unfortunate. Worse yet, on most other airplanes, the test airfoils were used directly, warts and all, because of NACA's mistaken assumtion that designers would synthesize "good" airfoils from the test data.

The GA ordinate tables are arranged with the "cusp" type airfoils at the top of each page,and the corresponding "no cusp" airfoils at the bottom of the page. The ordinate schedules generally follow the NACA format, however an additional station has been added at .25%C, to help define the leading edge. Unlike the NACA airfoils, GA airfoils do not have a true leading edge radius, and the leading edge is instead a curve defined by a series of discrete points.

Columns #2 and #3 of the ordinate tables are the thickness distribution (symmetrical shape) and the mean line, respectively, from which the three airfoils in each table are formed. Column #2 and #3 are listed for reference only. They are also usable for forming airfoils with thicknesses other than the three listed in the table. The procedure for this is to scale column #2 from 15% to the desired thickness, then add column #3 to obtain the desired airfoil.

E. Aerodynamic Characteristics of Wing Sections. The aerodynaminc characteristics of each 12% thick and 15% thick "no cusp" type GA wing section, as determined by the "Eppler" program,are given in Appendix IV, for Reynolds numbers of 2 million and 6 million, corresponding approximately to stall conditions and cruise conditions respectively for typical light airplanes. The characteristics of the 18% thick wing sections are not shown, but they are approximately the same as the 15% thick sections, with 2½ drag counts additional profile drag. See figure 16. Similarly, the characteristics of the "cusp" type sections are not shown, however they are approximately the same as the "no-cusp" shapes, with 2 drag counts less drag. See figure 14.

The characteristics at Reynolds numbers other than 2 million and 6 million may be estimated by interpolation and extrapolation, and by reference to figure 17.

The data in Appendix IV is for the smooth condition only. Figure 18 shows the effect of surface roughness on the performance of a typical GA airfoil, 37A315. The primary performance degradation is loss of laminar flow, causing a substantial increase in profile drag, and there is also some loss of maximum lift coefficient as well. The data is for roughness condition r=4, which is approximately equivalent to the NACA "standard roughness" condition used in their tests. According to reference 1, this is equivalent to a wing "well contaminated with bugs". However, the concensus is that the "standard roughness" imposed in the NACA tests is unusually severe, and is not likely to be encountered in actual aircraft service. Further, due to the superior leading edge shape of the GA airfoils, the performance degradation of GA airfoils due to surface roughness is probably not as severe as with NACA airfoils.

The performance data in Appendix IV is for wings without flaps. Although much experimental work has been done in wind tunnels with high-lift devices (see reference 1, chapter 8), there is no readily available computer program that can accurately predict airfoil section performance with flaps, due to the complicated flow patterns in the flap system. Accordingly, very little data is presented herein for GA wings with flaps, except for figure 2, which illustrates the principle that GA airfoils have significantly better flap effectiveness, that is, increase of maximum lift coefficient with a given flap system, due to the cumulative effect of the GA improvements, than their NACA counterpart airfoils. However, we offer the following as a "rule of thumb" for estimating the maximum section lift coefficient achievable with flaps, for GA airfoils. For large, well designed flaps, the greatest expectable factor of improvement in section Clmax for simple flaps is 1.5, for slotted flaps 2.0, and for Fowler flaps is 2.5. Thus, for a GA airfoil with a no-flap Clmax of 1.6, the Clmax with simple flaps could be as high as 2.4, slotted flaps as high as 3.2, and Fowler flaps as high as 4.0. These are section (2 dimensional) coefficients, and appropriate reduction must be made for three-dimensional effects. Also, they are the estimated maximum values for the best flap systems, whereas existing production flap systems probably fall well short of these numbers.

4. GA AIRFOIL SELECTION AND APPLICATIONS.

Figure 11 presents the matrix of the GA airfoils in this book, and offers some guidelines for selection of particular airfoils for given applications.

A. Effect of Series. Figure 13 shows the general performance differences between the turbulent GA 30 series and the laminar GA 35, 37, and 40 series. The GA 30 airfoils are the most conservative, and are appropriate for low-powered, Cub type airplanes. GA 35 airfoils are "entry level" laminar flow airfoils, and are suitable for airplanes like the Thorp T-18, at least when modestly powered as in the original design. GA 37 airfoils are general purpose laminar flow airfoils, to be used on a wide range of airplanes. GA 40 airfoils are for only the most sophisticated, high performance designs, such as the Glassair or White Lightning, where minimum drag at high speed is a strong requirement. They require the most accuracy in manufacture to realize their low drag potential. Also, the laminar buckets on the GA 40 airfoils are the narrowest of all the GA series, so the wing must be designed carefully to insure that the airplane is operating within the laminar bucket under all flight conditions.

B. Effect of Cusps. Figure 14 shows the effect of afterbody cusps on airfoil performance. Although both "cusp" and "no-cusp" airfoils are presented herein, for the great majority of applications the "no-cusp" airfoils are preferred. The main reason is to improve aileron effectiveness, as well as to simplify fabrication. The typical drag increase (.0002) is usually insignificant. For applications such as sailplanes and motor-sailers, where minimum drag is important, and especially with thicker airfoils, the "cusp" type airfoils can be used. However, with the "cusp" type airfoils, special "fat" aileron designs that have increased trailing edge angles should be used, to obtain acceptable aileron effectiveness.

C. Effect of Camber. Figure 15 shows the effect of different camber levels on GA airfoil performance. Of course, the main reason for using camber in the first place is to increase Clmax, but the resulting increase in Cm reduces top speed. Thus, it is a trade-off, and only as much camber should be used as necessary to meet the design requirements. See Addendum #1 for a discussion of the effects of increased Cm due to camber. In any event, large, effective flaps are the key to high aircraft performance, for they permit the use of lower cambered sections and reduced wing area, and this results in higher top speed. Accordingly, a good airplane design should start with the flaps, and then the rest of the airplane should be designed around the flaps. In general, the lower camber levels are used with higher powered airplanes and/or with large, effective flaps, while the higher camber levels are used with modestly powered airplanes, or with smaller or no flaps. Consideration must also be given to the design (cruise) lift coefficient of the airplane, to make sure that the bottom edge of the laminar bucket is below the Cldesign. For example, referring to figure 15, if the design lift coefficient is .25, the GA 37A415 could be used, but not the 37A615. This concern is especially important with thinner (12%thick) airfoils.

D. Effect of Thickness. Wing % thickness is primarily a structural consideration, however figure 16 shows some performance differences due to thickness. As stated above, the thinner airfoils have narrower laminar buckets, thus should be used cautiously. Furthermore, the stall tends to be less soft with the thinner airfoils, in spite of the "GA" improvents. Therefore, since the drag increase with increased thickness is so small, the minimum airfoil thickness to be used for most laminar flow applications is 15%. See Addendum #2 for a discussion of the effects of using airfoils that are too thin, especially at the wing tips.

E. Effect of Reynolds Number. The design Reynolds number can affect the choice of airfoil. For example, from figure 17 notice that the laminar bucket narrows considerably at higher Reynolds numbers, since laminar flow is harder to achieve as Reynolds number increases. Therefore, for higher Reynolds number applications, thicker airfoils and/or lower cambered airfoils may be necessary to insure that the airplane is operating within the bucket at top speed. Conversely, at low Reynolds numbers, Clmax drops off considerably, thus additional camber, and/or additional thickness, and/or more wing area may be needed compared to a higher Reynolds number application, for the same take-off/landing speed. For example, the BD-5 has a stall Reynolds number of less than 1 million, and uses a NACA 64-212 airfoil for the root section- a terrible choice for such a low Reynolds number, resulting in many needless fatal accidents. The airfoil should have been thicker, or more highly cambered, or better yet- both. See Addendum #4 below for a re-profiling scheme for BD-5 wings.

For very low Reynolds number applications, less than 1/2 million, such as in sailplanes, a phenomenon known as "laminar separation bubbles" followed by turbulent re-attachment near the end of the laminar run, may occur, on the top and/or the bottom surface, if the curvature of the wing surface is too great at that point. This results in increased drag. Any method to reduce the curvature of the wing surface at that point will help, however the usual fix is to use "turbulator tapes" span-wise slightly forward of the separation point to "trip" the flow to normal attached turbulent flow, avoiding the separation. This subject is beyond the scope of this book, and is normally not a problem for GA airfoils, which are forward loaded and thus have little camber in the afterbody, and this reduces the curvature on the airfoil surfaces in the afterbody compared to other sailplane airfoils. For this reason, "no-cusp" shapes are better than "cusp" type shapes regarding laminar separation bubbles. Also, the laminar separation bubble problem is magnified as the point of maximum airfoil thickness is moved farther aft, that is, beyond .40C, in which case radical cusps are required for satisfactory pressure recovery, and this leads to separation. "GA" airfoils, especially the no-cusp shapes, do not have this problem, for there are no GA airfoils with Tmax beyond .40C.

In summary, there are two ways to address the laminar separation bubble problem at low Reynolds numbers. The first way is to try to prevent the separation from occuring in the first place, with the basic airfoil shape. This requires thin, low cambered airfoils, with Tmax forward of .40C, with no cusps. However, this is not practical for sailplanes, for example, so the alternate solution is to go ahead and use the needed thick, high-cambered shapes with long laminar runs and short (cusp type) pressure recovery sections, and then force the transition to turbulent flow with turbulator tapes located top and bottom near the end of the laminar run, to keep the flow attached. The drag of the turbulator tapes is less than the drag that would otherwise occur from the laminar separation bubbles, so everyone is happy.

For exceedingly low Reynolds numbers, such as with very slow, indoor, small rubber band powered models, the flow is 100% laminar. To improve performance, a trip wire is often placed spanwise in front of the leading edge, to make sure that the flow over the entire wing is turbulent.

F. Applications. GA airfoils are currently being used on a wide range of applications from a Kolb "Firestar" ultralite to a high-performance Nick Jones "White Lightning" (both one-off, owner modified). Production kitplanes using GA airfoils include the Skystar "Kitfox" and the Ultravia "Pelican". All experimenters who have substituted GA airfoils for the original airfoils on their homebuilts have reported performance improvements, without exception. See figures 19 and 20 for typical applications.

5. AIRCRAFT PERFORMANCE PREDICTION

Figures 21 and 22 can be used in predicting aircraft performance. Computer predicted performance data is two-dimensional data,like wind tunnel data, that is, it is idealized data for a wing of infinite aspect ratio. The data may be corrected for three-dimensional effects, that is, corrected for wing aspect ratio, using methods described in standard aeronautical texts. For example, in determining a suitable angle of incidence for a particular airfoil on a given airplane, this usually means adding approximately ½ to 1 degree angle of attack to obtain the design lift coefficient (average) over the entire wing, depending on the aspect ratio. A higher aspect ratio needs less correction. For example- determine a suitable angle of incidence for a wing with an aspect ratio of 5 on an airplane with a wing loading of 15 lb/ft^2, and a cruising speed of 150 MPH, using the GA GA 37A315 airfoil. From figure 22 determine the two-dimensional design lift coefficient as Cldes= .25. From figure IV-13, determine the angle of attack necessary to obtain this lift coefficient as zero degrees. Now, since an aspect ratio of 5 is a fairly low number, add 1 degree for three-dimensional effects, and set the wing on the fuselage at an angle of incidence of +1 degree to the fuselage reference line. This method should be accurate enough for most cases.

As a double-check, note that the fourth digit in the "GA" airfoil designation (as in the NACA designation system for the 6-series airfoils) indicates the approximate design lift coefficient of the airfoil section, in tenths, that is, either .2, .3, .4, or .6. Thus, in our example with the GA 37A315 airfoil, the approximate design lift coefficient of this airfoil is .3, so this appears to fit well enough with the design lift coefficient needed for that particular airplane, that is, 0.25. If these numbers do not have an approximate match, the camber selection is probably not suitable for the application, or the estimated cruising speed of the airplane may be optimistic.

6. COMPUTER PROGRAM ACCURACY.

The computer program used to generate the predicted performance data of Appendix IV is the Eppler program "Profil", published in Fortran in reference 5, and adapted for personal computer use (MS-DOS) by reference 6. This is a simple program, and does not iterate on the boundary layer, thus it gives accurate results for only well-designed airfoils such as "GA" airfoils, that have smooth, continuous mean lines. For poorly designed airfoils with discontinuities in the mean line, such as the NACA 23012 airfoil, it gives inaccurate results, especially for Clmax, because it cannot follow the complicated effects of premature separation bubbles at higher angles of attack common in those airfoils. But who wants to use poorly-designed airfoils anyway?

A comparison of computer results and wind tunnel data for a typical airfoil, NACA 64-215, is shown on figure 23. Note that the accuracy of the computer data is quite good, all things considered, and is a credit to Dr. Eppler. The program appears to be especially accurate in predicting laminar/turbulent phenomena, separation bubbles, etc. However, the program appears to overstate drag by at least 10%, and to overstate lift by about 5%. The worst disagreement appears to be in the pitching moment coefficient, which the program overstates by at least 30%. For this reason, as absolute data, the computer results should be used cautiously. However, for comparative purposes, such as shown on figure 2, the program is excellent.

The program appears to be calibrated for best accuracy for airfoil thicknesses of about 15%. The program shows a considerable loss of Clmax for thinner (12% thick) airfoils, and a gain of Clmax for thicker (18% thick) airfoils, as shown by figure 16. This does not agree with wind tunnel data, which typically shows negligible differences of Clmax as a function of airfoil thickness. Dr. Eppler comments on this in his book, reference (9), and suggests that, due to the limitations of wind tunnel testing, especially the change in the ratio of airfoil cross-section area to the tunnel throat size with different airfoil thicknesses, perhaps his computer data is more accurate than wind tunnel data. In any event, we suggest that when comparing the performance of different airfoils with the computer program, one should use the same airfoil thickness for both airfoils being compared.

The Eppler program has a routine for simulating the performance of an airfoil with simple flaps. However, the predicted performance is wildly optimistic compared to wind tunnel data, because the program of necessity, when the flap chord and deflection are inputted, draws a completely new, idealized airfoil that is unrealistically smoothed so that the nemesis of the program (premature flow separation) will not occur. Furthermore, the program cannot begin to simulate the complicated flow patterns of more elaborate flap systems such as slotted and Fowler flaps. Thus, we did not spend much time with the flap routine in this program. In spite of this shortcoming, however, the flap routine is valuable for predicting relative differences of flap performance between similar airfoils, as shown on figure 2. Another use for the flap routine is to predict Cm vs Clmax for a given flap system. In other words, if a simple flap system yields a Clmax of 3.0 by the program, then the reported Cm will be reasonably accurate for a flap system with that Clmax, even though it may take a considerably more elaborate flap configuration to achieve thatClmax in practice.

The Eppler program also does not include any correction for compressibility effects, treating the airflow as being incompressible. However, this effect is so minor as to be insignificant, for the flight regimes of typical light general aviation air planes. For example, the wing loading of a Piper Cub, 7#/ft^2, yields an average pressure difference of only 7 #/ft^2/144 = .05 psi between the top and bottom surfaces. Compared to the atmospheric pressure of 14.7 psia, this amounts to a pressure difference of only 0.3%. For higher performance airplanes, such as a Bonanza with a wing loading of 20 #/ft^2, the difference is still less than 1%. Thus, the assumption that the flow is incompressible is not bad.

The Eppler program for performance analysis calculates velocity ratios across the airfoil at each angle of attack, top and bottom surfaces, then prints these velocity ratios (point velocity divided by free stream velocity) in tabular and graphic form. Alternatively, it will calculate and print out pressure ratios across the airfoil. Then, the program calculates boundary layer data for each angle of attack at specified Reynolds numbers, including coefficients of lift, drag, and pitching moment, the turbulent area, and the separated area. These values are also printed out in tabular form, and in a "boundary layer summary plot". A sample analysis print-out is shown on figures 24-1 thru 24-4.

Several years ago Dr. Eppler brought out an "improved" version of his program. However, the new program seems to understate section drag, and also reports laminar bucket width as increasing with an increase in Reynolds number, which is not the case. Further, the results appear to be overly sensitive to data point (ordinate) accuracy, thus I still favor the old program.

Both programs include routines for design of airfoils by the modern, one-step "inverse" method, whereby idealized velocity diagrams are first calculated, and then airfoil top and bottom surface ordinates are calculated from the velocity data. That is OK, but the possibility exists of "creating a monster", such as the very recent NASA NLF(1)-0115F airfoil, discussed in Addendum No. 5 below. This is less likely to happen with the "classic" method used by NACA and "GA Airfoils". In any event, neither the airplane, the wind tunnel, nor the computer knows or cares which method was used to design the airfoil, and the final performance figures are the only thing that counts. Thus both methods are valid. Any airfoil, even if hand-sketched, can be broken down into a symmetrical thickness distribution and a camber profile (mean line), and much can be learned about the airfoil by inspecting and analizing these two components separately.

7. GA CUFFS FOR WING REWORK

As noted earlier, all of the NACA airfoils have poor nose profiles, due to the faulty NACA design methods.The result is poor slow-speed performance, in many cases. The NACA 4-digit (turbulent) and NACA 6-series (laminar) airfoils can be improved by adding a leading edge cuff to the airfoil, which drops the center of the leading edge approximately ½% to 1% of chord length. The method for designing these "drooped leading edges", shown on figure 25 for the NACA 64-212 airfoil, is to design a new mean line forward of 10%C to obtain at least 12 degrees of initial mean line slope, necessary for a soft stall. Then, using the existing top surface and the revised mean line, we calculate the ordinates for the bottom surface.

The performance analysis for this cuffed airfoil shows a dramatic improvement in the stall compared to the original 64-212, as well as a slightly wider laminar bucket- see figure 26. Also shown is the performance of an upper surface modification proposed by Hicks et al at NASA Ames. This cures the sharp stall, but it also reverts the airfoil to a turbulent section, causing high drag. The performance of the GA 37-212 is also shown, for comparison.

This cuff modification should be used only as a "quick fix" on existing wings, not for new construction- use a GA airfoil instead. Also, it does not work on the NACA 5-digit airfoils, like the 23012, which already have too much leading edge droop, and a different stall mechanism, than the 4-digit and 6-series airfoils. See Addendum #3 below.

8. MISCELLANEOUS NASA AIRFOILS

With the coming of the space age, NACA's name was changed to NASA, and the agency's responsibilities were expanded to include national space projects as well as the old-line aeronautical projects. The total budget is currently about 14 billion dollars yearly, of which about 1.2 billion is related to "aeronautics" projects, a fairly constant amount yearly. However, over the past several decades, work related to General Aviation projects has dwindled to nearly zero, due to "budget restraints", in NASA's words. The only general aviation airfoil work, for example, has been the sporadic release of several random airfoils. These have usually been the brain-children of particular NASA individuals, rather than any concerted effort, and have been disappointments.

A. NASA GAW-1. In 1974, the NASA GAW-1 and GAW-2 "Whitcomb" airfoils (later designated LS(1)-0417 and LS(1)-0413) were released. These airfoils are characterized by a large leading edge radius, and a slab-shaped profile having the maximum thickness at .40C. This produces a long laminar run, at the expense of laminar bucket width. The leading edge has a droop of about .75%C, more than it needs to give the airfoil its soft stall characteristic. The low profile drag is enhanced by the pronounced cusp in the afterbody, and the blunt trailing edge. The worst feature by far, however, is the aft-loaded camber profile- an outrageous mistake for an airfoil that was touted to be a general aviation panacea. This gives the airfoil a pitching moment coefficient three times as high as it needs to be for the amount of camber in the airfoil, and this produces very high trim drag, rendering the airfoil unsuitable for general aviation use. In effect, the airfoil has two notches of flap permanently built into it. Unbelievably, NASA chose this airfoil for their "ATLIT" project, described below.

Figure 27 shows the bizarre press release concerning the NASA "ATLIT" (Advanced Technology Light Twin) project in 1974, which turned out to be a disaster, performance-wise. The project was doomed to failure from the beginningdue to the selection of the high-Cm NASA GAW-1 airfoil for the wing. The predicted performance figures were no doubt calculated without regard to the high Cm of the airfoil, a major blunder. The spoilers for roll control were also unsatisfactory, experiencing control reversal at low and medium deflections. Nothing new was learned by using a longer span, tapered wing- these effects had been known for decades. This merely contaminated the data, making it impossible to compare to the base-line (factory) airplane. NASA was so embarrassed by the poor performance of this airplane that the project engineer never even bothered to write a final report on the project, but he got promoted anyway. He now has a top position at NASA Langley. In retrospect, NASA forgot that their function is to conduct basic aeronautical R&D, not to "show the industry that they are wrong, and this is the proper way to build a GA light twin". This is an example of what can happen if R&D is not market driven, but is directed by irresponsible bureaucrats. The airplane was bought back by Piper for salvage of engines, instruments, etc, and was then sold to a technical school for students to drill holes in it, with the proviso that it never fly again.

Figure 28 shows the NASA "ATLIT" airplane during wind tunnel tests at NASA Langley. This is your money.

The GAW-1 airfoil was also used on the Beech Skipper and the Piper Tomahawk, with disappointing results on both airplanes, particularly at high speed.

A common rework for this airfoil is to fill in the bottom side cusp, as shown on figure 6. This reduces the aft loading somewhat, reducing Cm and trim drag, but the airfoil remains a mish-mash.

B. NASA NLF(1)-0215F Airfoil. This airfoil, due to Somers, and described in Reference 7, was designed originally as a sailplane airfoil, thus it has too much camber in it for typical powered airplane applications. Accordingly, the trailing edge must be reflexed approximately 10 degrees as shown on figure 29, for powered airplanes. Even with the reflex, high aileron hinge moments (stiff ailerons) remain, so a common rework is to make the ailerons and flaps flat-bottomed, as shown on figure 29, to salvage the airfoil. Of course, the better solution to the problem is simply to design an airfoil with less camber in it in the first place.

Another strange feature of this airfoil is the camber "dip" at .60C, as shown on figure 3. This is an attempt to move the transition point aft, extending the laminar run to reduce profile drag, which it does. However, the fallacy here is that the negative camber at .60C causes negative lift, and this causes more (induced) drag than is saved by the extended laminar run, reducing overall airfoil efficiency.
All in all, NACA/NASA has never reached a concensus as to the best shape for camber profiles for general aviation airfoils. See the "GA" camber profiles in Appendix II for a comparison.

C. NASA NLF(1)-0414F Airfoil. This airfoil, figure 30, due to Viken, was envisioned as a very low drag laminar flow airfoil for high-powered, high performance general aviation airplanes. The point of maximum thickness on the airfoil is at .45C, and the maximum camber, centrally loaded, is approximately 2.5%c. This makes the airfoil comparable to the NACA 66-414 airfoil. Unlike the NACA airfoil, however, the NLF airfoil has enough leading edge droop, approximately .35%C, to give the airfoil decent slow speed performance.

Realizing that the price paid for locating the maximum thickness so far aft is a very narrow laminar bucket, NASA flight tested the airfoil on a Cessna 210 with narrow chord (12%C) "cruise flaps", reflexed upwards approximately 7 degrees at cruise, and adjustable. This was an effort to shift the laminar bucket vertically as needed, to keep the airplane "in the bucket" under various flight conditions, that is, at various typical weights and speeds encountered in aircraft service. This proved to be difficult and impractical, and the net wing drag turned out to be more than anticipated. The scheme was thus considered to be unsatisfactory for the intended application. This is an example of a "single design point" airfoil.

Notice from figure 30 that the -7 degree cruise flap setting, necessary to reduce the maximum camber from 2½%C to 1%C for cruise, creates a region of negative camber at .85C. This causes a drag penalty, similar to what we have seen previously from reflexing the trailing edge of the NLF(1)-0215 airfoil (combined with the camber dip at .60C), reducing the efficiency of the airfoil. As before, the better choice would be to design the airfoil with less camber (NLF(1)-0214), to keep the airfoil clean at cruise- in short, a conventioal design approach. However, the airfoil would still be a single point design. Accordingly, our conclusion is that the point of maximum airfoil thickness should be no further aft than .40C, for typical general aviation applications.

This airfoil has been modified, however, by Team Nemesis, and is being used successfully on the Nemesis Formula I racer. This is a very special application, a "single point" airplane, and this supports our conclusion that the airfoil is unsatisfactory for most general aviation applications.

It is interesting to note that NASA's Cessna 210 flight test program for this airfoil, reported in NASA TM 85788, was similar to the poorly designed "ATLIT" flight test program described above. On the Cessna 210, the span was increased from 39 ft. to 42ft, making it impossible to compare the performance of the new airfoil to the "base line" airfoil. The inescapable conclusion is that NASA was here again trying to show that they can design airplanes better than the prime manufacturer, rather than being satisfied with doing legitimate research work. At least, in both cases, NASA proved how not to build a general aviation airplane, and this is worth something- I guess.

9. RELATED SUBJECTS

A. Blunt Trailing Edges. All GA airfoils have been designed with sharp trailing edges, for uniformity. However, sharp-edged airfoils, on wings and control surfaces, work better if they are cut off as square as possible at approximately 99% of chord. The reason for this is that the trailing edges have finite trailing edge angles, and this causes the flow to separate before reaching the trailing edge anyway. So you might as well cut it off. The airfoil will still act as if it were 100% long.

B. Flying Wing Airplanes. Due to the lack of a tail, flying wing airplanes must use zero-Cm airfoils, thus GA airfoils are not suitable. Further, due to pitch stability concerns, flaps cannot be used on flying wings. Thus, due to these two design constraints, the performance of any flying wing airplane will be mediocre compared to an equally sophisticated conventional airplane.

C. Canard Airplanes. This configuration has similar design constraints. Flaps cannot be used on the main wing. Further, the forward surface must always be more heavily loaded than the main wing, so that the canard will always stall first. Since the main wing can never be allowed to stall, it never reaches its maximum lift coefficient, and is thus relatively inefficient. Therefore, due to these two design constraints (no flaps, no-stall main wing), the canard configuration can never be as efficient as an equally sophisticated conventional configuration. Further, since full-stall landings are not possible, landing speeds are relatively higher than for comparable conventional airplanes.

D. Three-Surface Airplanes. Control surfaces placed in the tail have longer moment arms than those placed in the nose, thus are more effective. So why bother with control surfaces in the nose?

E. Leading Edge Devices, such as slots and slats. Thin airfoils, and other airfoils that are subject to leading edge separation, can benefit from these devices. GA airfoils, however, are soft-stall airfoils, which means that leading edge separation is not a problem, hence slots and slats cannot offer much improvement. There are two exceptions, however. First, in the case of movable slats that move out from the leading edge and increase the wing area, more lift will result simply from the increase of wing area. Secondly, leading edge slats contribute a positive (nose up) pitching moment that can help offset the large negative pitching moment due to large trailing edge flaps. In spite of these potential advantages of leading edge devices, with GA airfoils, the primary effort at lift augmentation should be with effective trailing edge flaps.

F. Case Study, Rutan Canards. Figure 31 shows that Rutan used the GAW-1 airfoil scaled up to 18% thickness for the Vari-Eze. The GAW-1 airfoil, however, with its high Cm, has a considerable center of pressure travel, contributing to pitch stability problems. Accordingly, for the successor Long-Eze, Rutan chose a forward loaded (lower Cm, lower C.P. travel) airfoil to improve pitch stability. However, The Long-Eze airfoil is a turbulent airfoil- why? With such a large, lightly loaded main wing, the natural choice seems to be a low Cm, soft stall laminar flow wing. I don't have an answer to this puzzle.

Postscript

As stated in the Introduction above, the original NACA airfoils were never intended for actual airplane use, and, accordingly, "GA Airfoils" has been written to address this need. In retrospect, one must ask why NASA, in the forty-odd years since the NACA airfoil work was done, has made no effort similar to "GA Airfoils"? How much valuable time and effort has been lost, and how many precious lives have been needlessly wasted, because NASA failed to address this problem? They certainly must have recognized the need, so why didn't they act? Why did they fail to put an end to the myth that the NACA airfoils were actual airplane airfoils, and why didn't they admit that NACA's mistakes and omissions needed correction?

It is hard to think of an answer to these questions without becoming cynical. One possible explanation is that NASA doesn't know how to do the job. That is hard to believe, however, in view of the astronomical budget that NASA is blessed with each year to pay for talent and facilities.

Another possible explanation is that, by the immediate post-WWII era, the United States had achieved undisputed world-wide supremacy in aviation affairs, in spite of the "good advice" from NACA, and we enjoy this number one position to this day. Accordingly, NASA could have felt that there was little need to do additional aviation R&D work, nor to review the old work, so they "rested on their oars". So then the question arises, why wasn't NASA's funding cut off at that point? And why has their funding continued yearly to this day? After all, the agency was set up originally to subsidize an infant industry, which aeronautics was in the 1920's. But U.S. aviation is now not only a mature industry, but it enjoys a runaway position of world supremacy! To continue to subsidize this industry with federal funds, at the expense of starving deserving infant industries of the 1990's, makes no sense whatever.

Furthermore, a more basic, philosophical concern exists. In a free society, one must always be on guard against excessive central planning, in which decisions regarding allocation of production resources are made by bureaucratic guesstimates, rather than economic realities. R&D work is a legitimate production expense, and falls in this same category, and must be held in check by the realities of market economics. For this reason, we must always be suspicious of government R&D as opposed to private sector R&D. Since the U.S. aviation industry is now undoubtedly mature, they are unquestionably able to do their own R&D work, in the normal course of their business. For this reason, the aviation R&D budget of NASA should be terminated immediately. And let's not be mislead by those who would argue that it is necessary to continue NASA's aviation R&D to insure that the U.S. aviation industry maintains its position of leadership.That is a specious argument, and ignores the realities of limited federal funds. Hard choices have to be made in Washington, but fortunately this is an easy one. Further, let's not confuse the issue by claiming that NASA's work is vital for national defense. The military can conduct their own R&D, under their own budget. Accountability for government spending insists on this.

No one will be hurt if the NASA aviation R&D shop is closed, and the U.S. Taxpayer will be eternally grateful. In fact, now is a good time to terminate this Federal jobs program. The currently booming air transport industry can easily absorb the "out-placed" personnel resulting from "down-sizing" NASA. Now is the time to act. Aviation R&D should be market-driven by Adam Smith's "invisible hand" of free enterprise, not subject to politics and bureaucratic whims.

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APPENDIX III

AIRFOIL ORDINATES

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4 — 4-DIGIT (TURBULENT) AIRFOILS

5 — 5-DIGIT (TURBULENT, ZERO C_m) AIRFOILS

6 — 6-SERIES LAMINAR FLOW AIRFOILS

FIGURE 1

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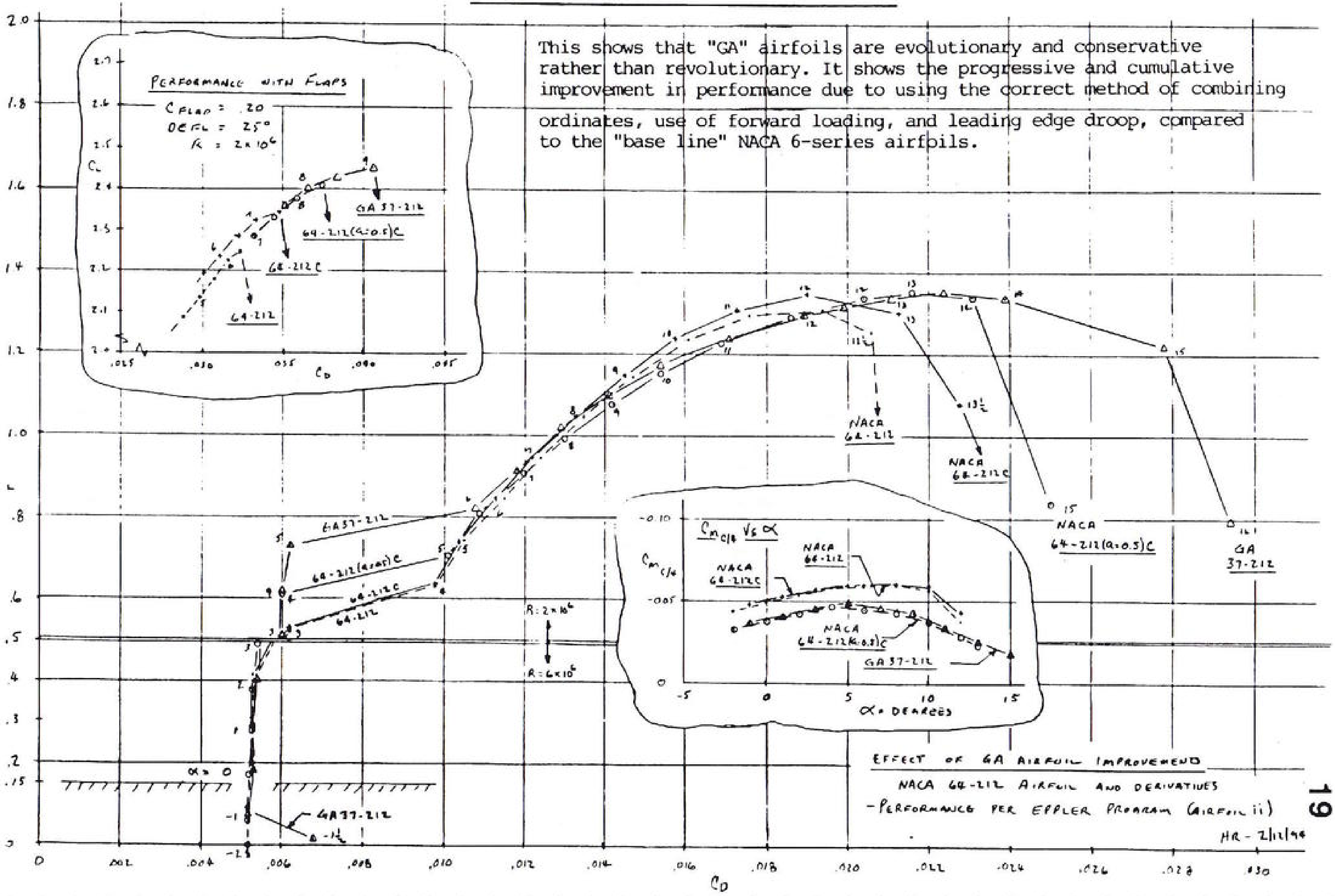
- "NO CUSP" AIRFOILS

— LOW C_m (FORWARD LOADED) AIRFOILS

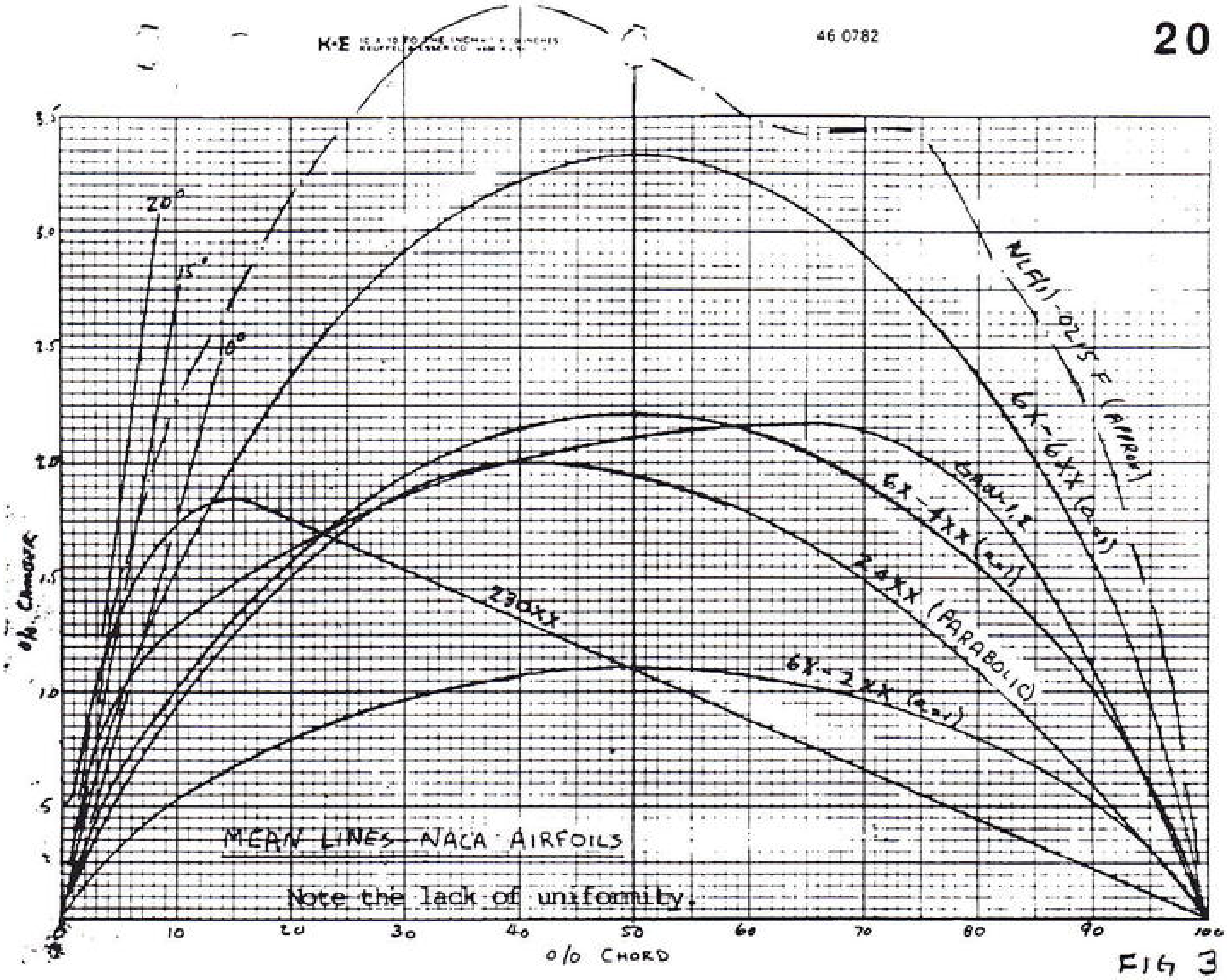
EVOLUTION OF GA AIRFOILS

This shows that "GA" airfoils are evolutionary and conservative rather than revolutionary. It shows the progressive and cumulative improvement in performance due to using the correct method of combining ordinates, use of forward loading, and leading edge droop, compared to the "base line" NACA 6-series airfoils.



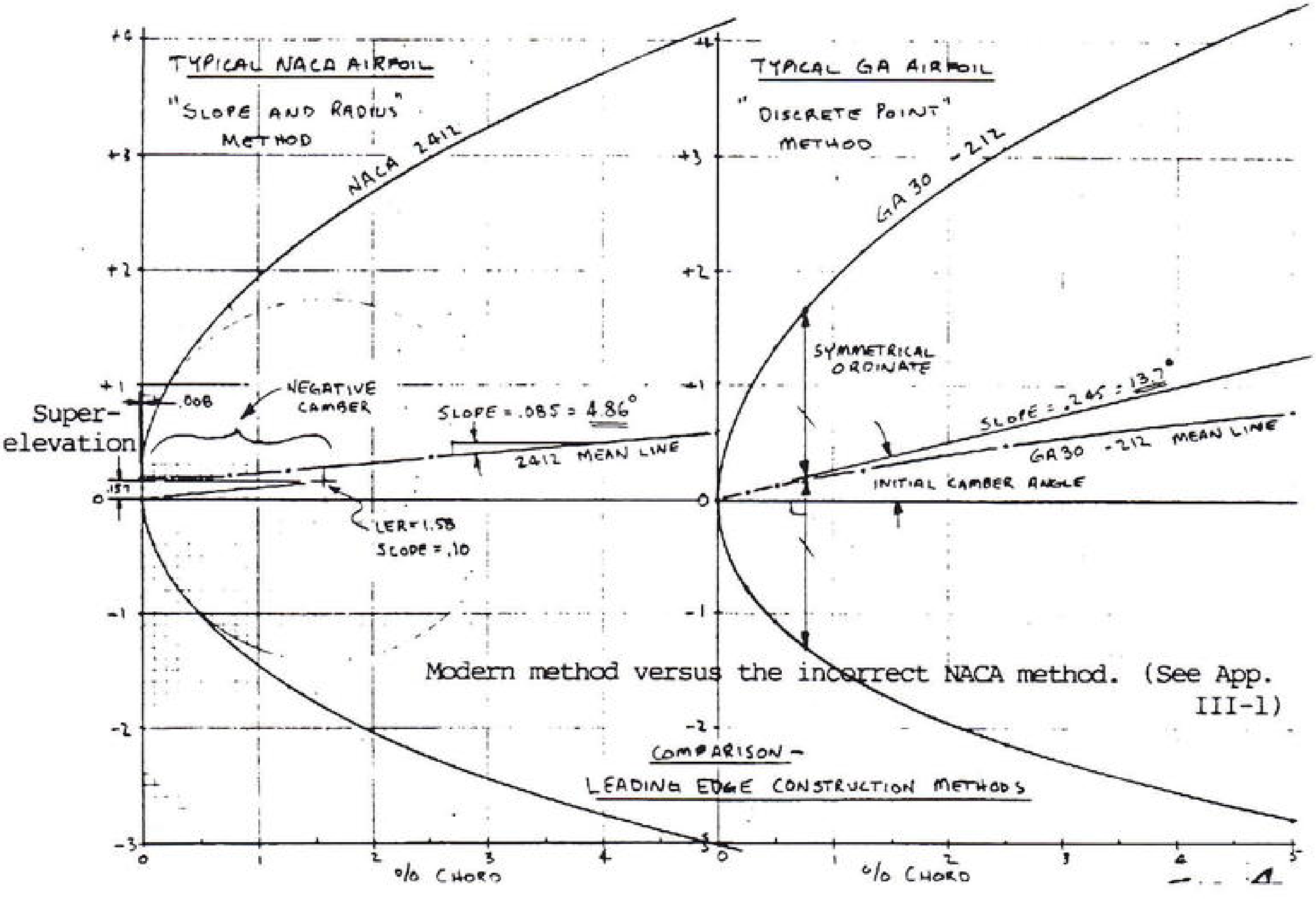
EFFECT OF GA AIRFOIL IMPROVEMENT
NACA 64-212 AIRFOIL AND DERIVATIVES
- PERFORMANCE PER EPPLER PROGRAM (AIRFOIL ii)

HR - 2/12/94

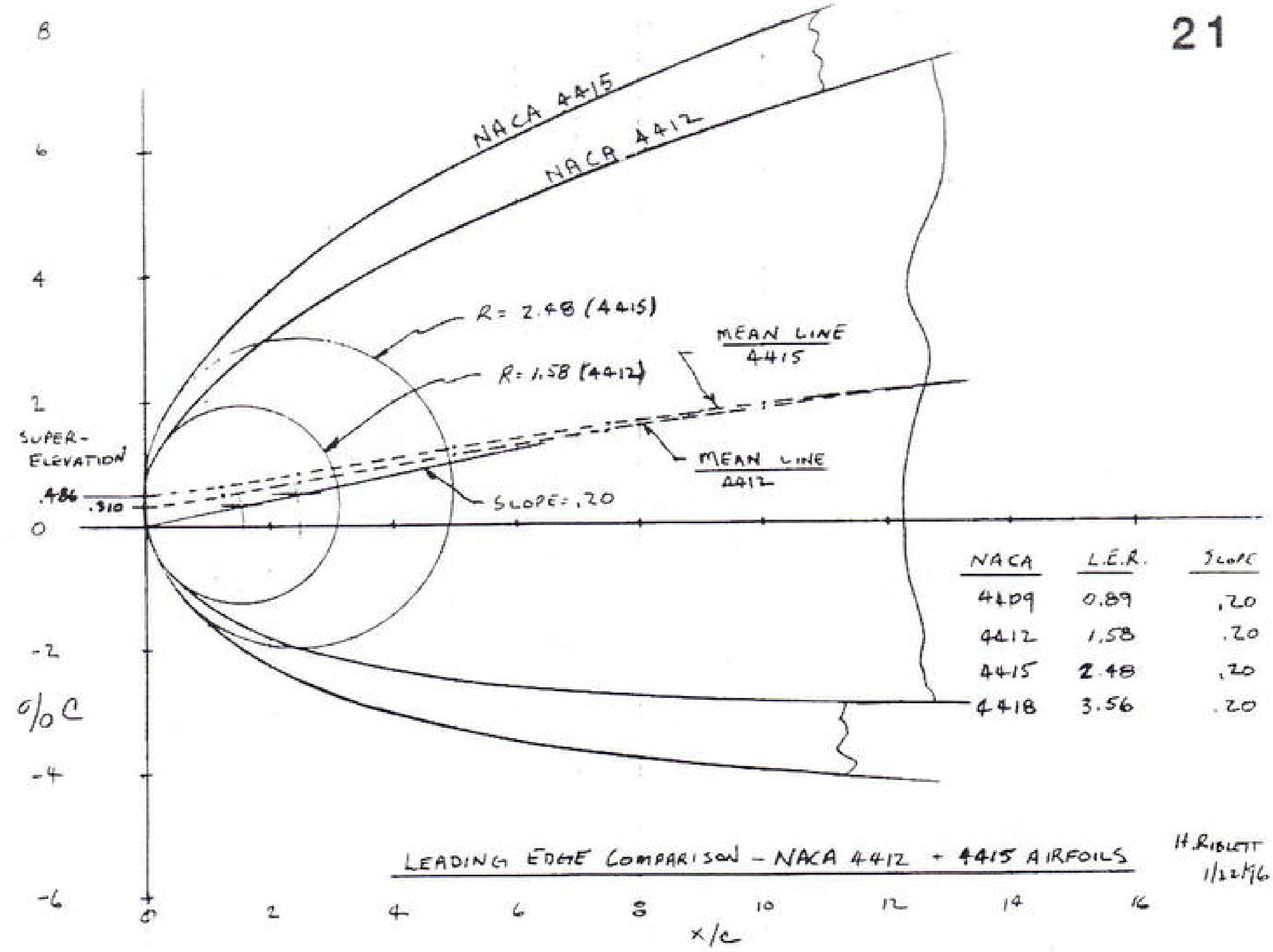


Note the lack of uniformity.

FIG 3



Modern method versus the incorrect NACA method. (See App. III-1)



NACA	L.E.R.	SLOPE
4409	0.89	.20
4412	1.58	.20
4415	2.48	.20
4418	3.56	.20

LEADING EDGE COMPARISON – NACA 4412 + 4415 AIRFOILS

H. RIBLETT 1/22/96

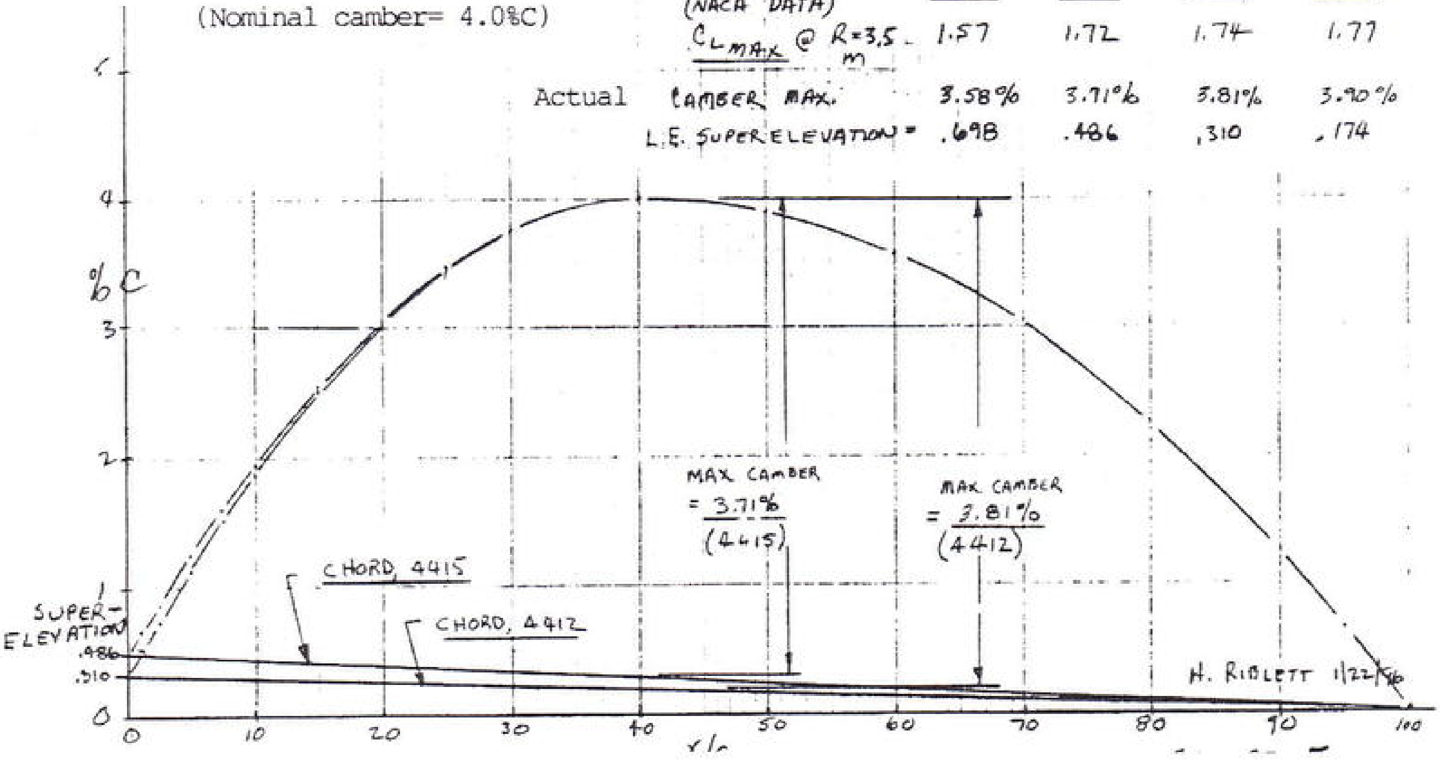
This shows that the amount of camber in NACA airfoils varies with thickness, due to the faulty "slope and radius" method of leading edge design. Therefor, NACA's test data regarding effect of thickness on airfoil performance is invalid, or at least contaminated.

CAMBER COMPARISON

NACA 44XX AIRFOILS

(Nominal camber= 4.0%C)

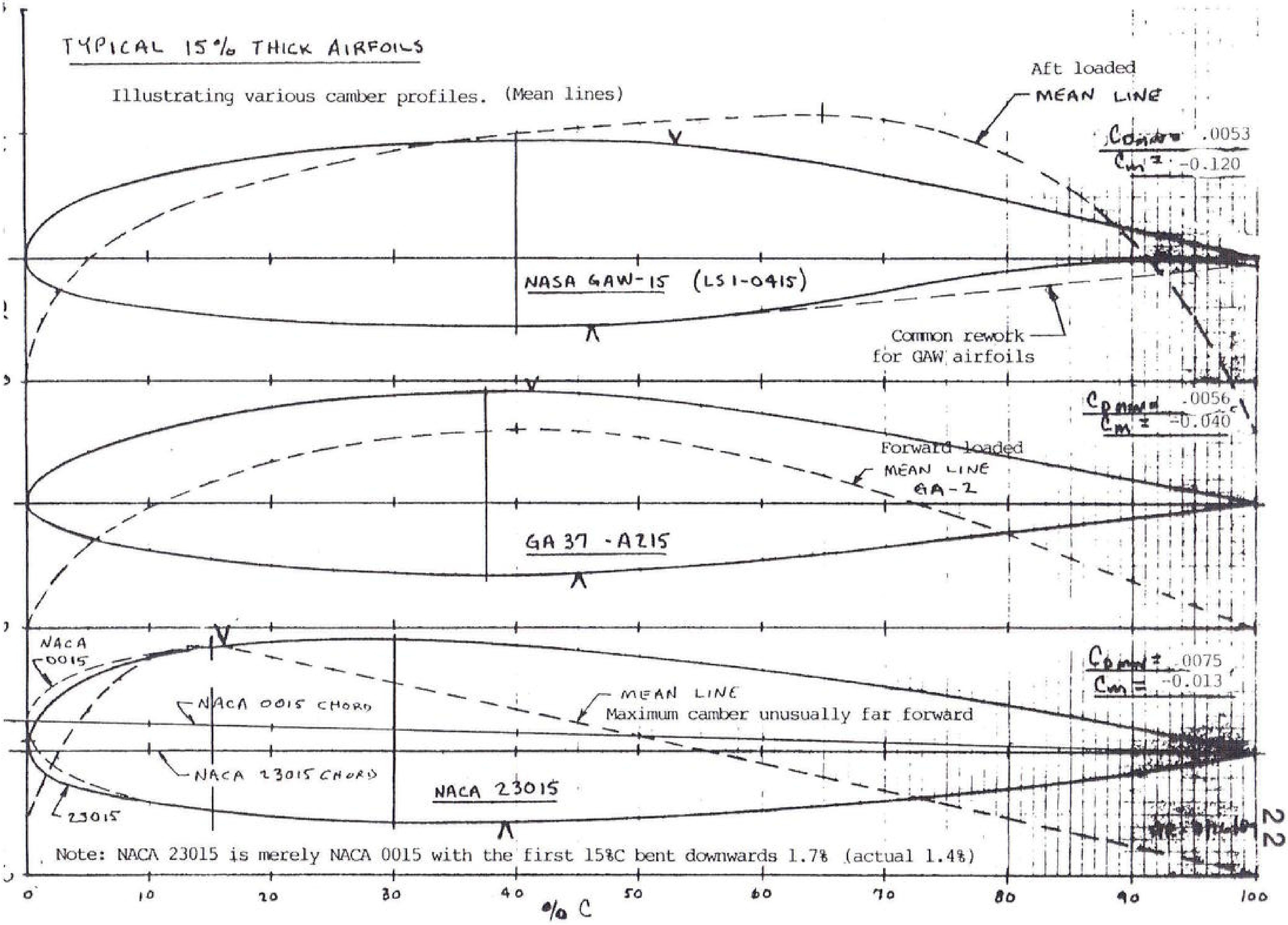
		4418	4415	4412	4409
(NACA DATA)	$C_{L_{MAX}}$ @ R_m=3.5	1.57	1.72	1.74	1.77
Actual	CAMBER MAX.	3.58%	3.71%	3.81%	3.90%
	L.E. SUPERELEVATION =	.698	.486	.310	.174



H. RIBLETT 1/22/96

TYPICAL 15% THICK AIRFOILS

Illustrating various camber profiles. (Mean lines)


Aft loaded
MEAN LINE
C_Dmin = .0053
C_m = -0.120
NASA GAW-15 (LS1-0415)
Common rework
for GAW airfoils
C_Dmin = .0056
C_m = -0.040
Forward loaded
MEAN LINE
GA-2
GA 37 -A215
C_Dmin = .0075
C_m = -0.013
NACA
0015
NACA 0015 CHORD
MEAN LINE
Maximum camber unusually far forward
NACA 23015 CHORD
NACA 23015
23015
0
10
20
30
40
50
60
70
80
90
100
% C

Note: NACA 23015 is merely NACA 0015 with the first 15%C bent downwards 1.7% (actual 1.4%)

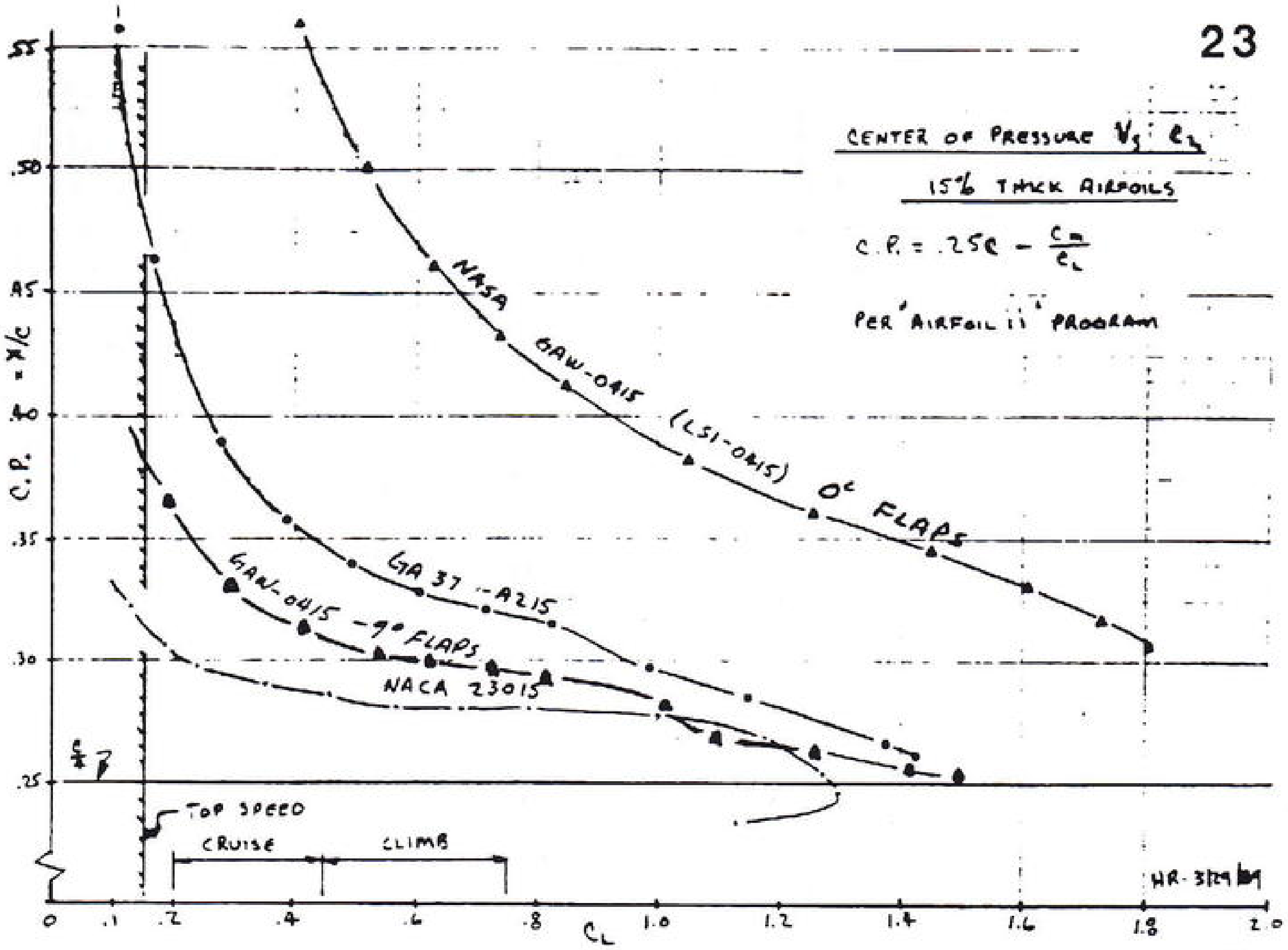
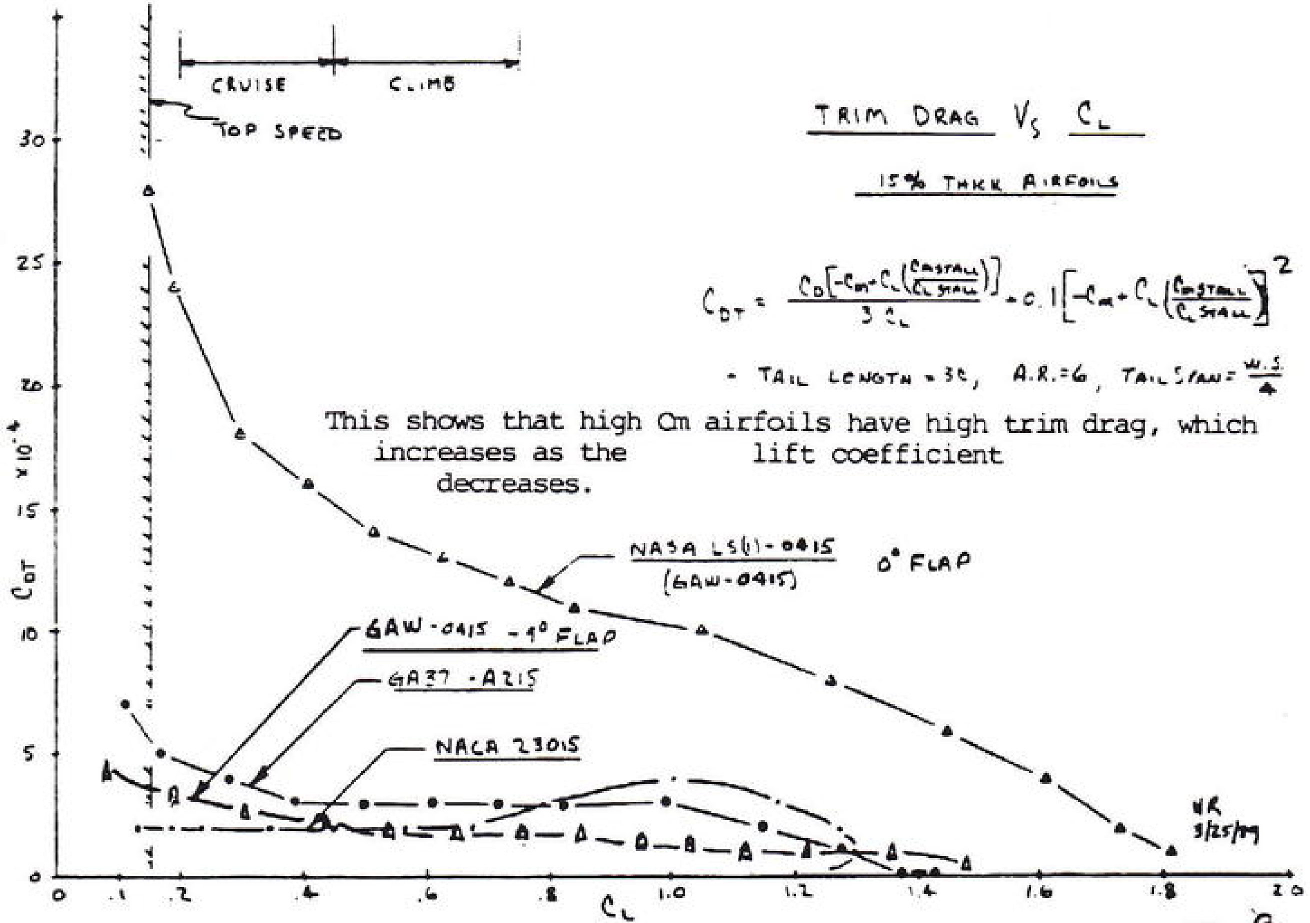
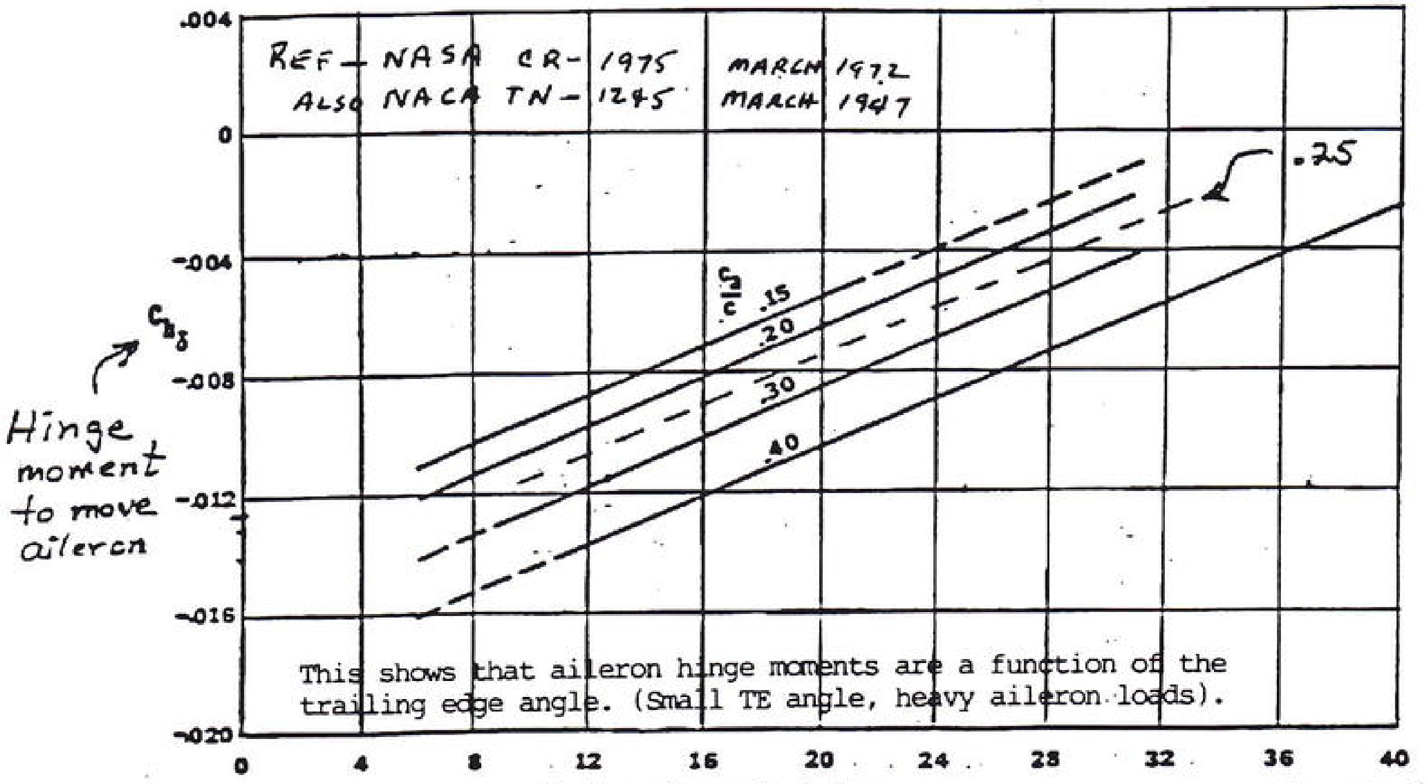


FIGURE 7



This shows that high Cm airfoils have high trim drag, which increases as the lift coefficient decreases.

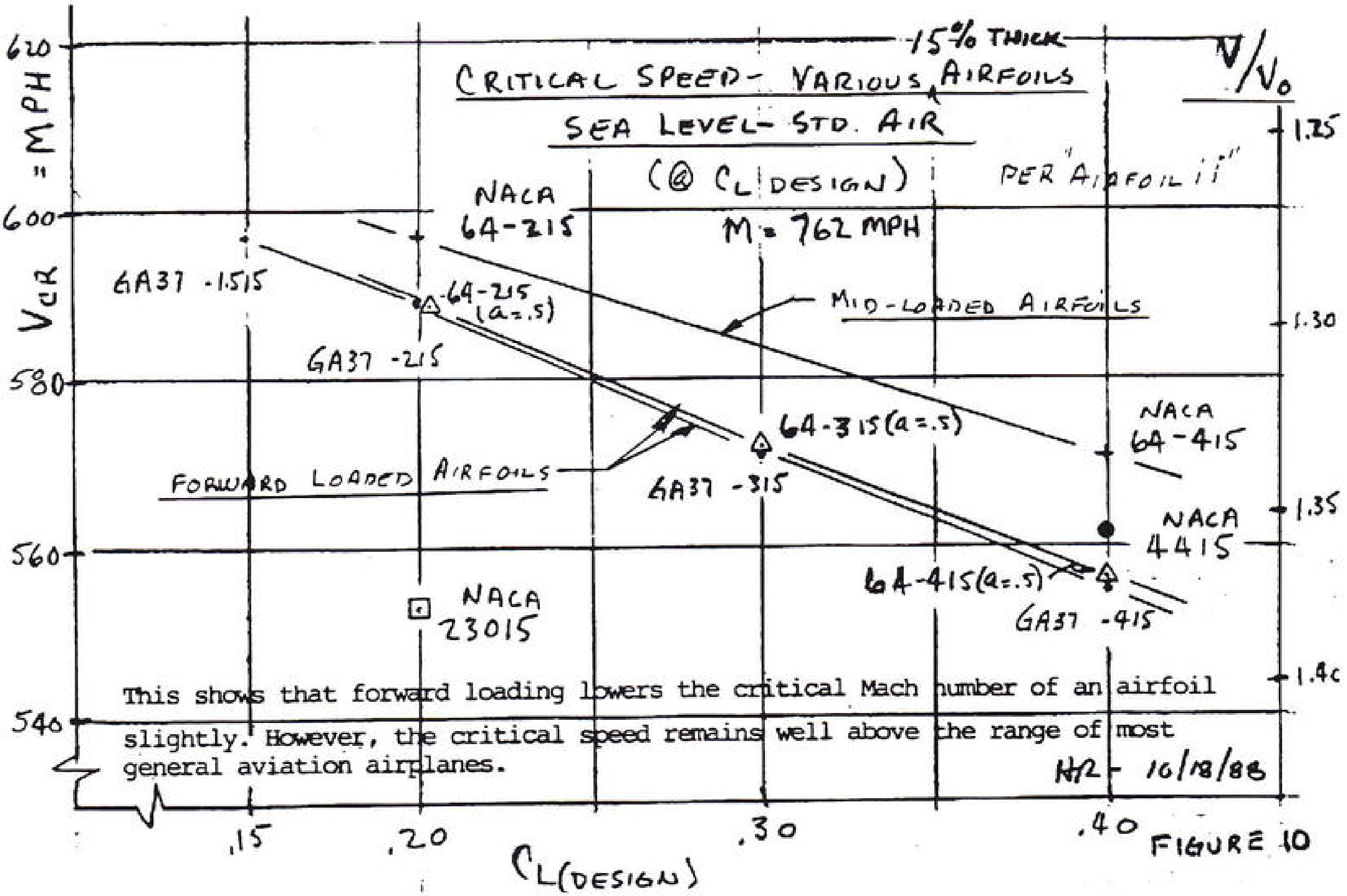
FIGURE 8



This shows that aileron hinge moments are a function of the trailing edge angle. (Small TE angle, heavy aileron loads).

FIGURE 9

Hinge moment parameters of plain 2-D ailerons with various chords and values of φ. Gaps sealed; c_a/c=aileron to wing chord ratio.



This shows that forward loading lowers the critical Mach number of an airfoil slightly. However, the critical speed remains well above the range of most general aviation airplanes.

HR - 10/18/88

FIGURE 10

AIRFOIL SELECTION CHART (GA AIRFOIL MATRIX)

	GA 30		GA 35		GA 37		GA 40	
	- SIMPLE AIRCRAFT - TURBULENT FLOW - t_{max} @ .30 x/c		- GEN. PURPOSE - LAMINAR FLOW - t_{max} @ .35 x/c		- MED. HIGH PERFOR. - LAMINAR FLOW - t_{max} @ .375 x/c		- VERY HIGH PERFOR. - LAMINAR FLOW - t_{max} @ .40 x/c	
	—	A	—	A	—	A	—	A
	ST. TAIL	CONVEX TAIL	CUSP	NO CUSP	CUSP	NO CUSP	CUSP	NO CUSP
LOW CAMBER	212 215 218		212 215 218		212 215 218		212 215 218	
MEDIUM CAMBER	312 315 318		312 315 318		312 315 318		312 315 318	
MED. HIGH CAMBER	412 415 418		412 415 418		412 415 418		412 415 418	
HIGH CAMBER	612 613.5 615 616.5 618 (FIG III-19)		612 615 618		612 615 618		612 615 618	

Example- "GA 35A315" - General Purpose, Laminar flow, No cusp, Med. camber, 15% th

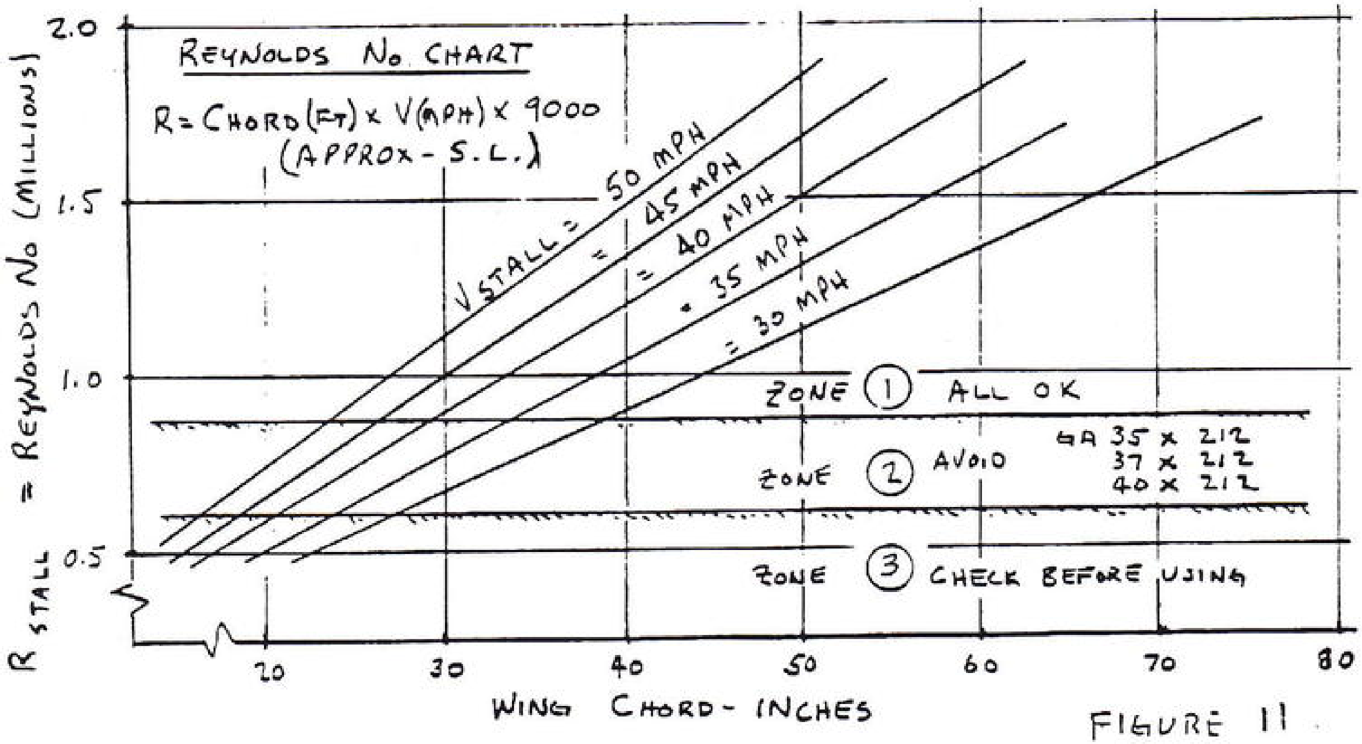
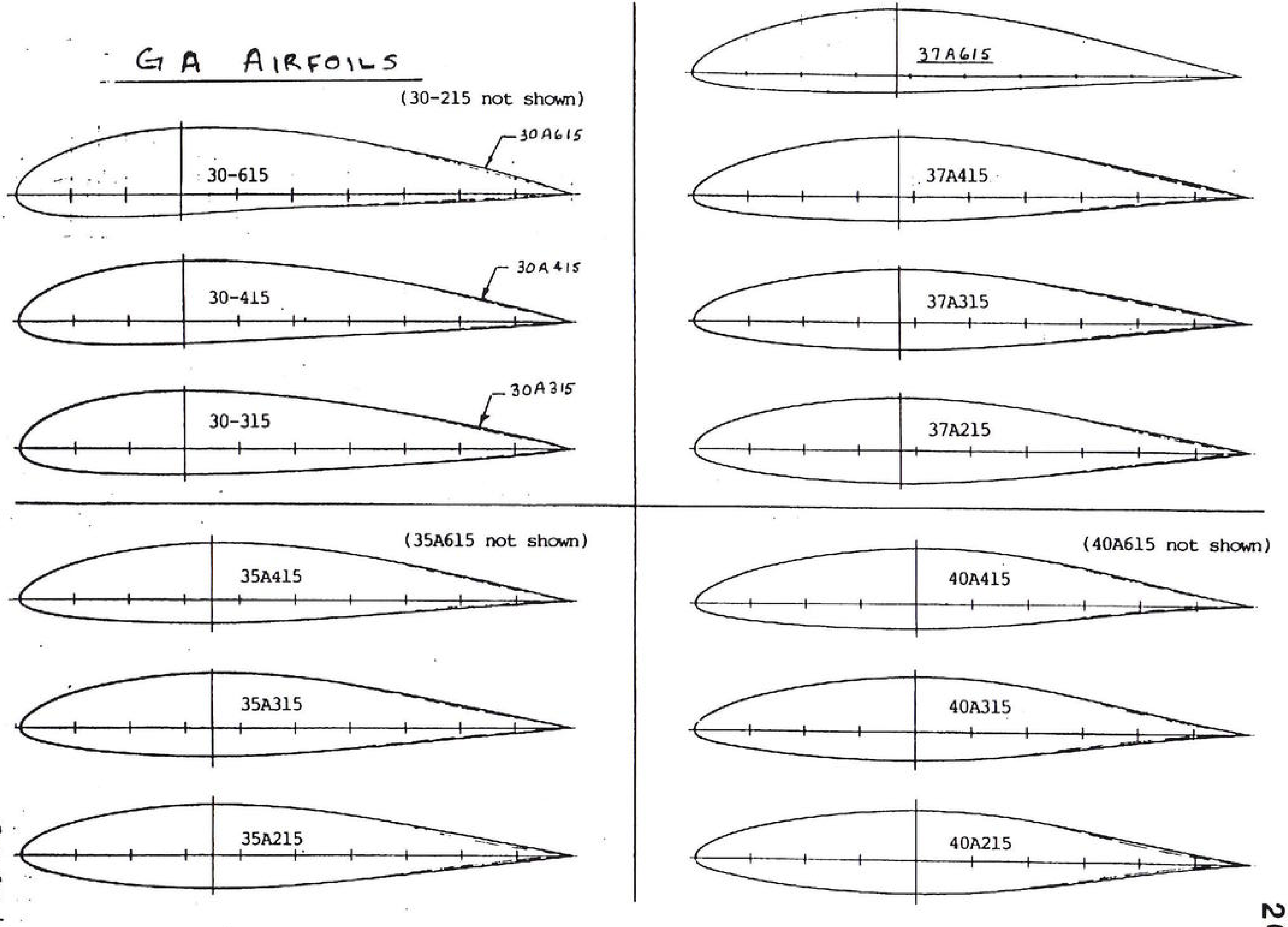


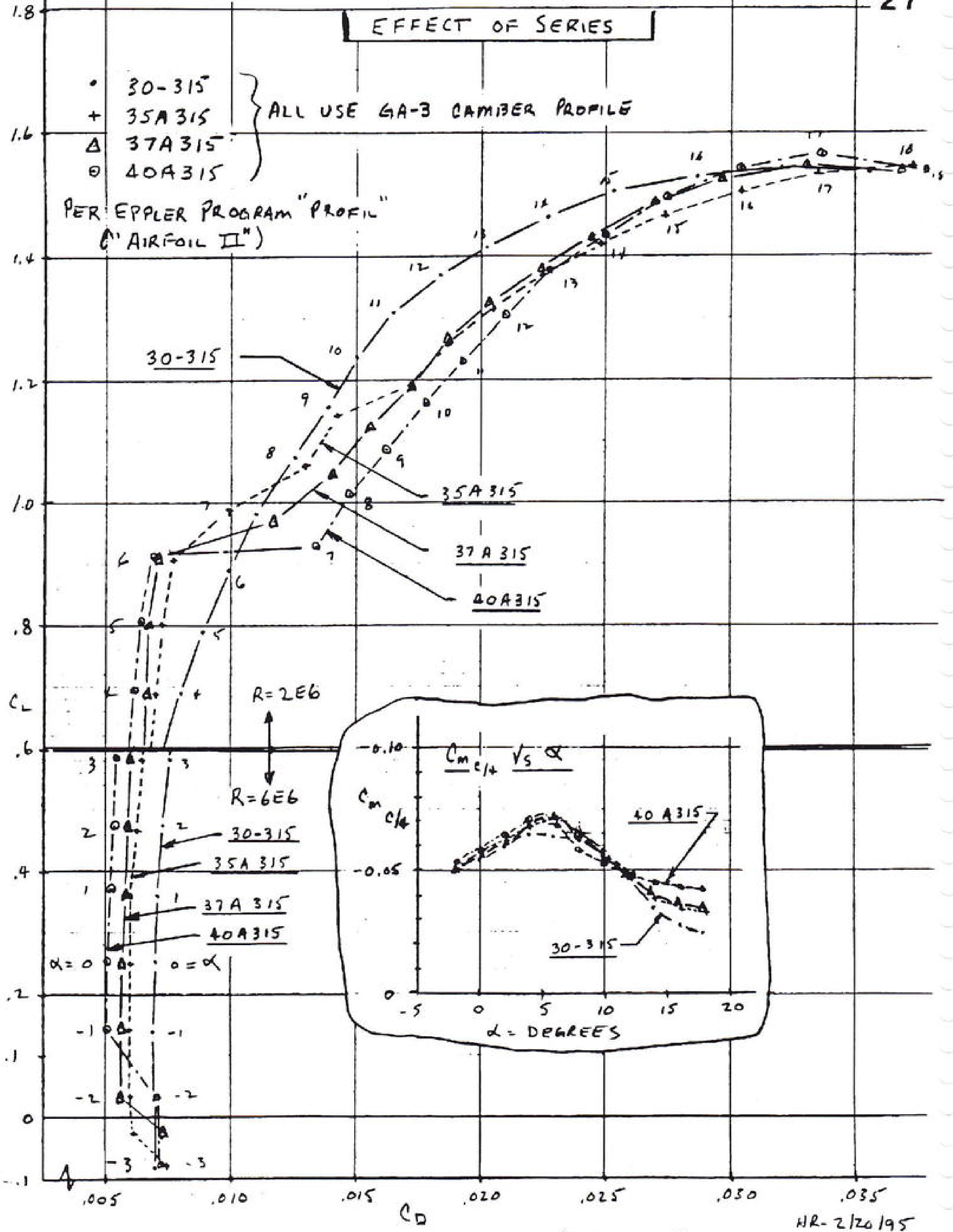
FIGURE 11

GA AIRFOILS


(30-215 not shown)
30A615
30-615
30A415
30-415
30A315
30-315
37A615
37A415
37A315
37A215
(35A615 not shown)
35A415
35A315
35A215
(40A615 not shown)
40A415
40A315
40A215

15% Thick GA Airfoils

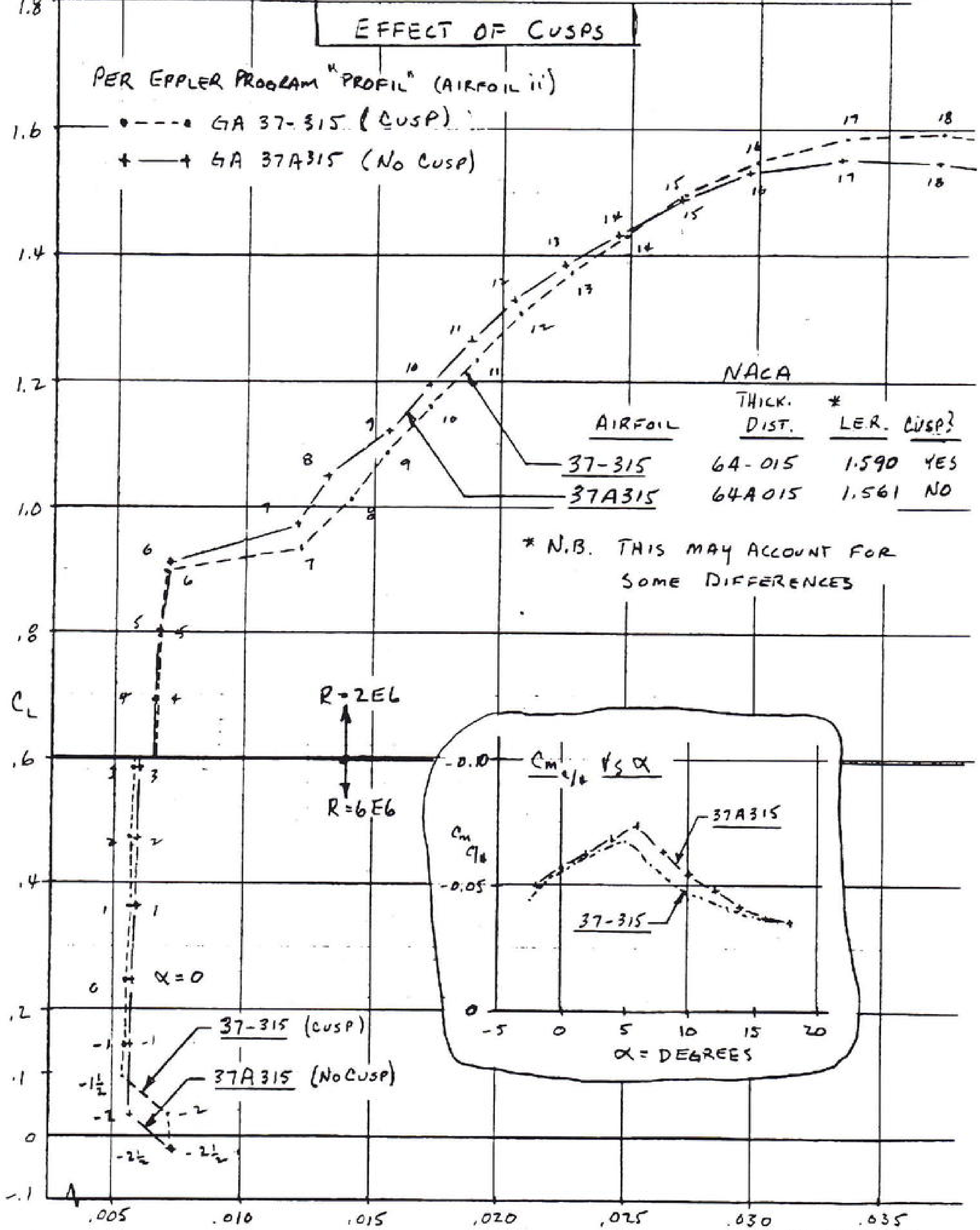
FIGURE 12



HR-2/20/95

GA XX-315 AIRFOILS - VARIOUS SERIES

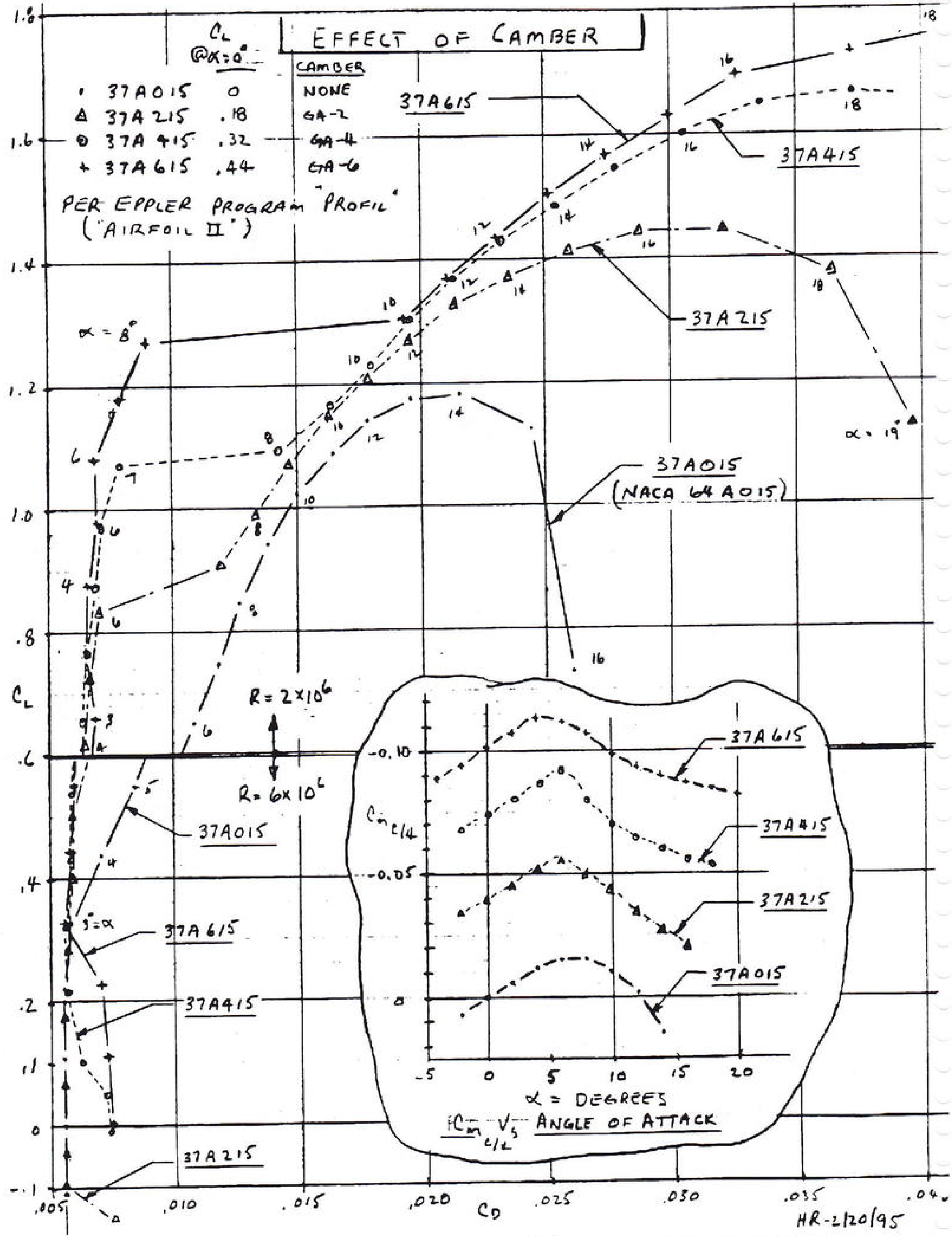
FIG 13



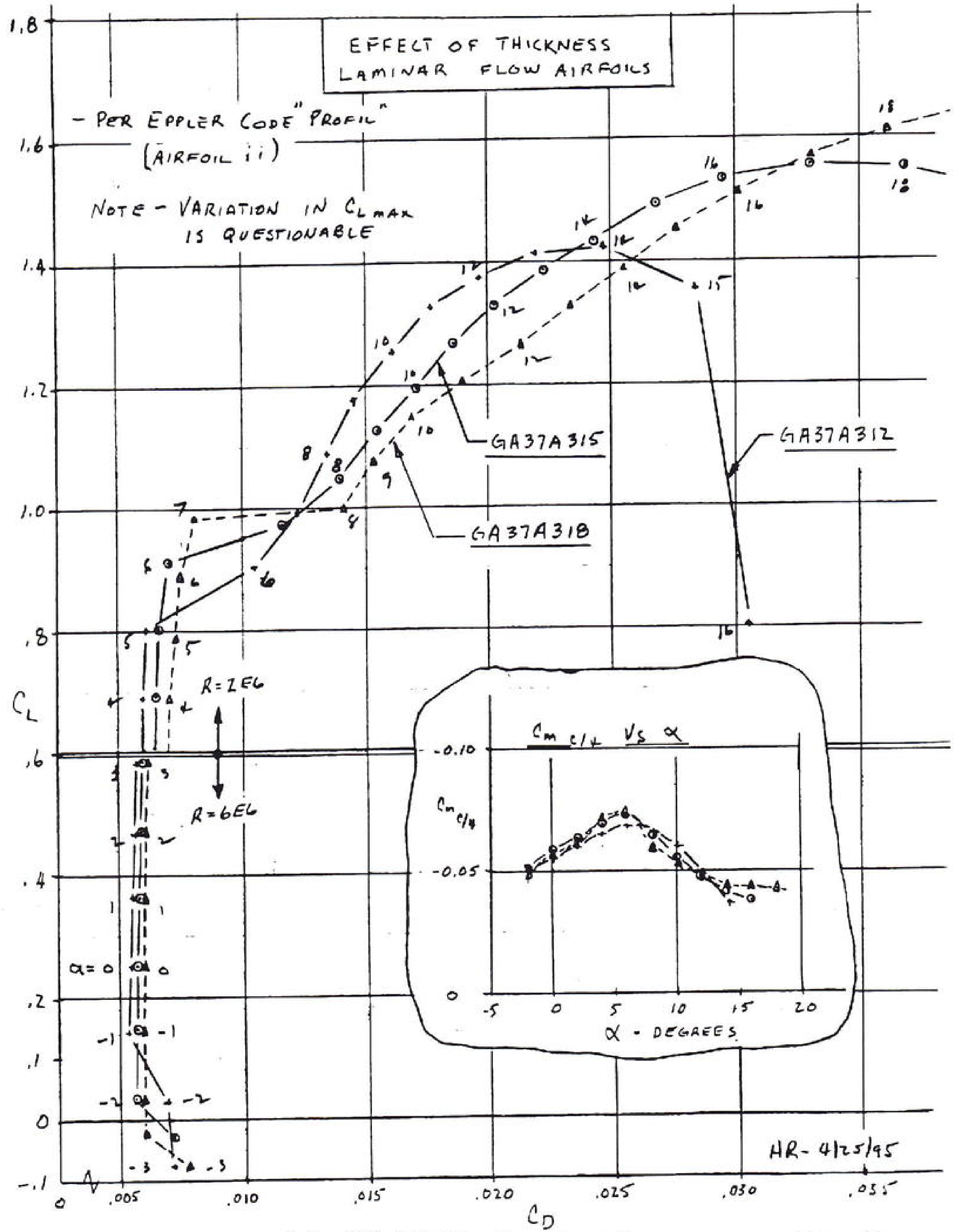
AIRFOIL	NACA THICK. DIST.	* L.E.R.	CUSP?
37-315	64-015	1.590	YES
37A315	64A015	1.561	NO

* N.B. THIS MAY ACCOUNT FOR SOME DIFFERENCES

HR - 2/26/95



GA 37A X15 AIRFOILS AT VARIOUS CAMBER LEVELS FIG 15



EFFECT OF THICKNESS
LAMINAR FLOW AIRFOILS

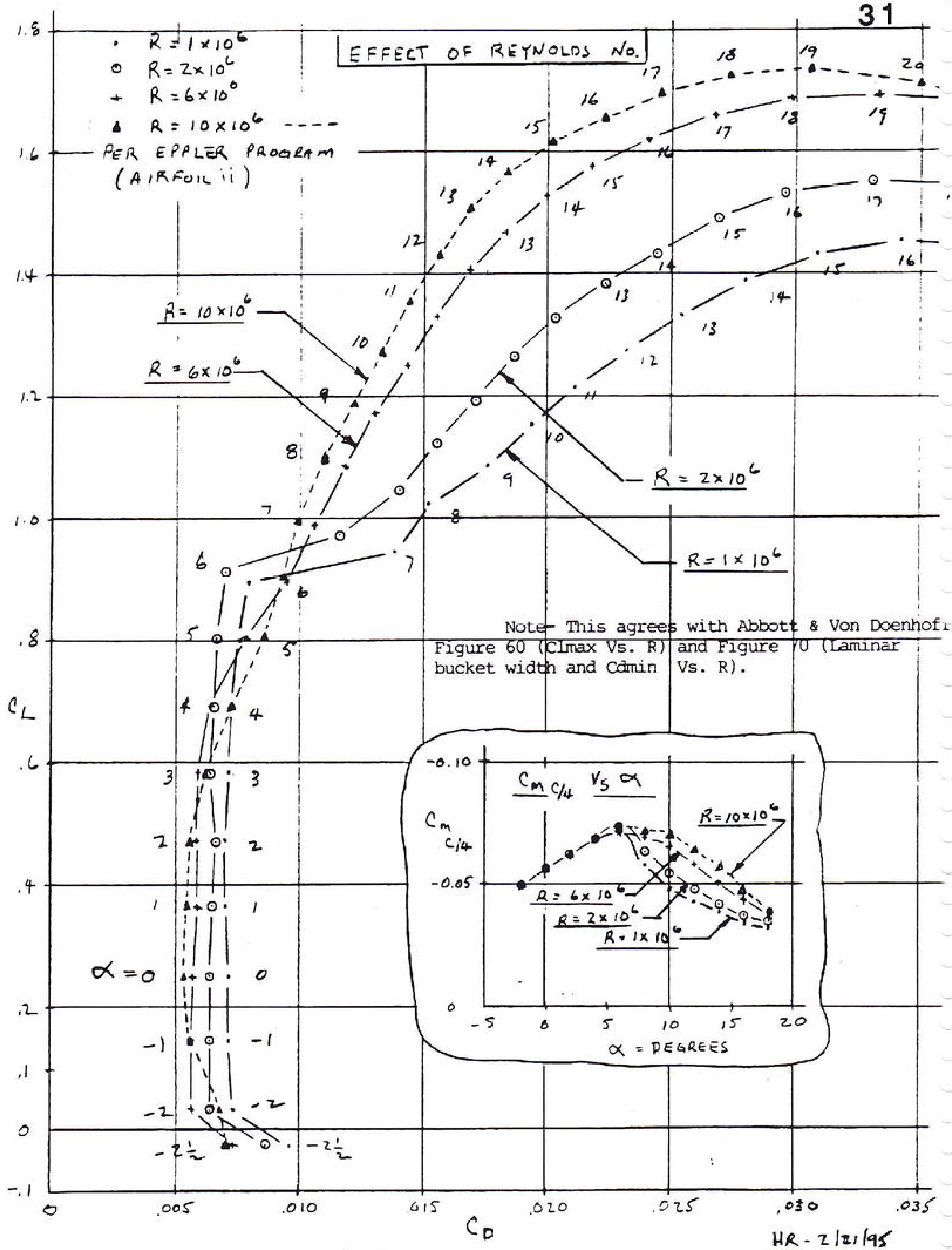
- PER EPPLER CODE "PROFIL"
(AIRFOIL II)

NOTE - VARIATION IN $C_{L\,MAX}$ IS QUESTIONABLE

HR- 4/25/95

GA 37A31X AIRFOILS

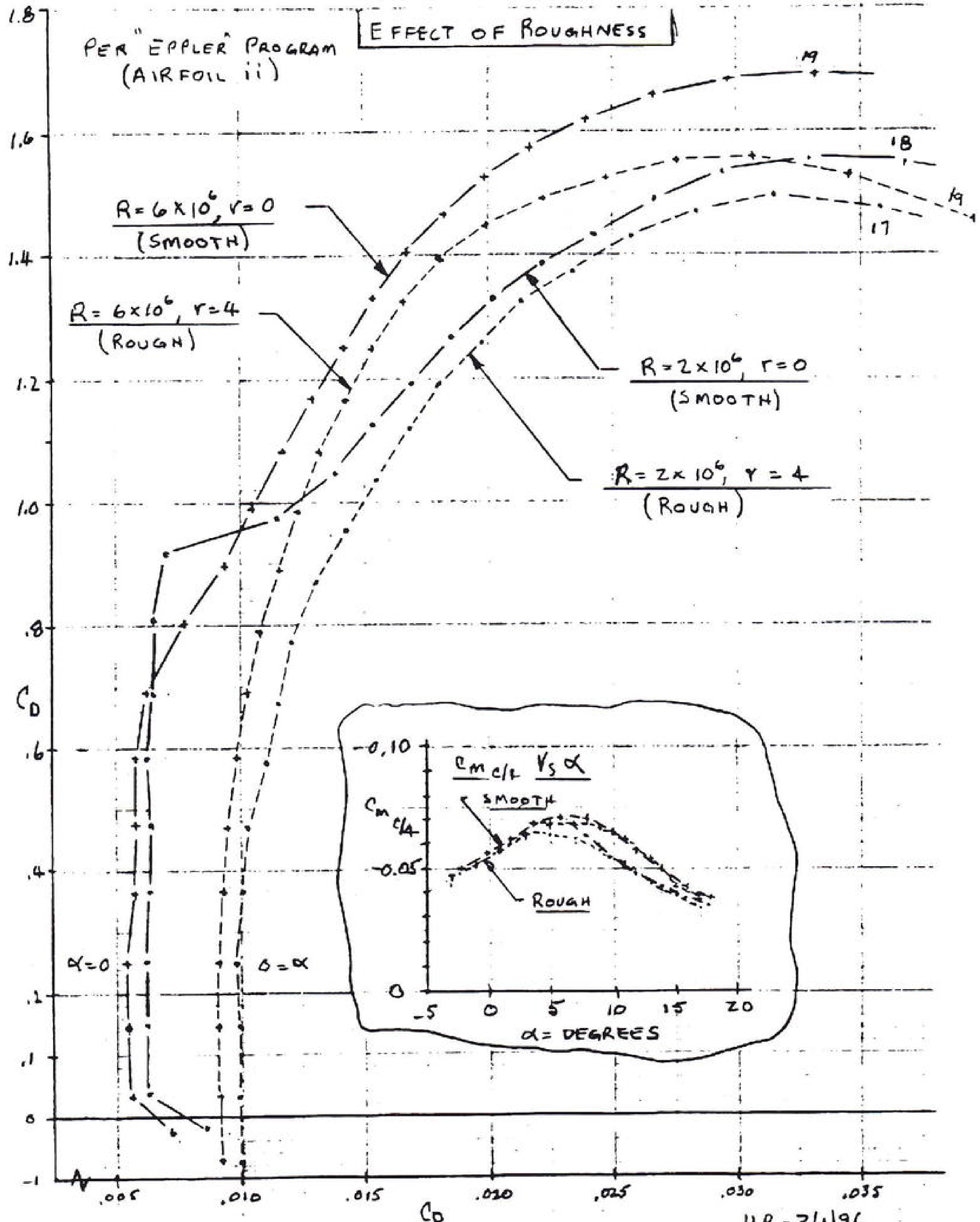
FIG 16



Note- This agrees with Abbott & Von Doenhof
Figure 60 (Clmax Vs. R) and Figure 70 (Laminar
bucket width and Cdmin Vs. R).

GA 37A 315 AIRFOIL

HR - 2/21/95
FIG 17

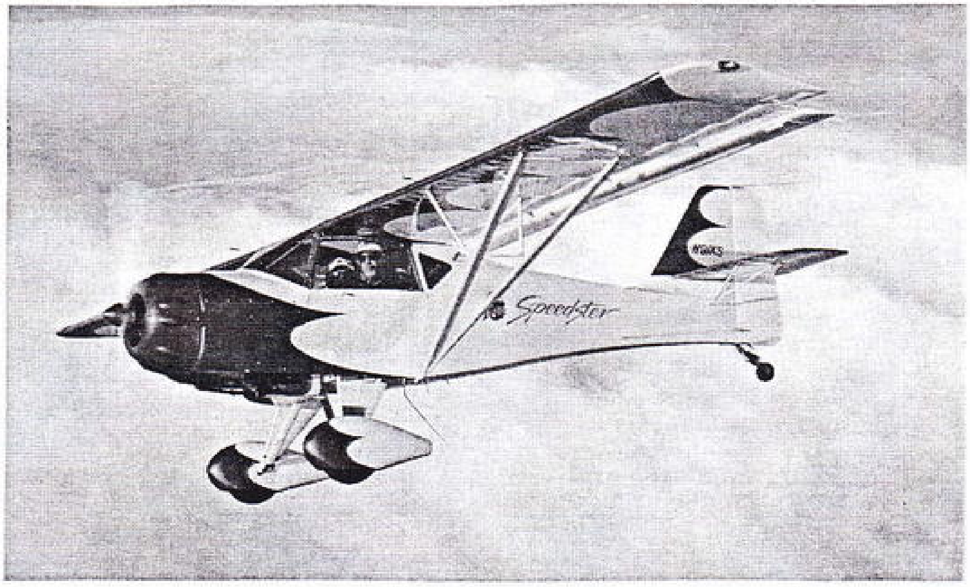


HR-2/1/96

GA 37 A 315 AIRFOIL

FIGURE 10

KITFOX



KITFOX SPEEDSTER

With a look reminiscent of the racing planes of the 30's, the Kitfox Speedster **struts its stuff**. Favorable aerodynamic characteristics and an 80 hp Rotax 912 engine give it top performance. Classic looks and leading edge engineering allows a top speed of 5 times its normal stall speed, offering maximum efficiency and safety. The Speedster represents real sport performance and value at a price you can afford. World renown aerobatic pilot Jim Franklin demonstrated the Speedster at OshKosh '92 and gave it rave reviews, calling it "A great little airplane, I want one!"

GA
30U-612

PELICAN

ULTRAVIA AERO INTERNATIONAL INC.

300-D Airport Rd., Mascouche, Quebec, Canada J7K 3C1

Dear Mr. Riblett,

I am indeed using the GA 30U-315 airfoil on the Pelican « PL » and we are very happy with it.

Best regards,

Jean René

Jean-René Lepage,
P.Eng.
President

SONERAI

FLETCHER BURNS REPORTS ON HIS SONERAI II WITH RIBLETT AIRFOIL

From the Sonerai Newsletter

(REPLACES NACA 64-212)

The wings are thicker than normal after a bent wing on my way to Oshkosh (which is another story). I came up with a 15% thicker airfoil from Harry Riblett and I love it. I don't think it slowed me down at all. I've sandbagged them down to 9 G's static without a wrinkle. Thicker is stronger! This is a general aviation airfoil not a pylon race airfoil. It lands much softer now, climbs better with two aboard and the roll rate really improved (although it didn't need to).

GA 37A315

JAN 1995

EAA

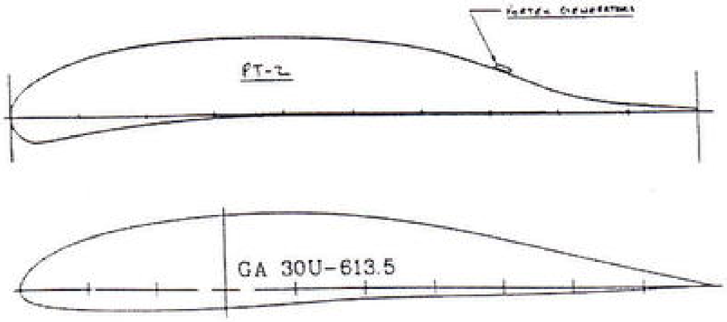
FIGURE 19

A New Wing for the Protech PT-2 Airplane

Harry Riblett
416 Riblett Lane
Wilmington DE 19808
(302) 994 0479

In January of 1993 Ken Rogers called me and told me that he was disappointed with the performance of his Subaru Legacy powered Protech PT-2 (see Contact! issue #11 for engine installation details). In this same issue of Contact! was an article that I had written describing on of my airfoils, GA 30U-613.5. This airfoil is designed to be a replacement for high-lift "Cub" type applications and Ken wondered if it might not be a good choice for the PT-2. I assured him that it would be, but suggested that since his airplane was complete and flying, there may be something that we could do to "patch up" the old wing. Accordingly, Ken sent the details of the old wing to me, and I made a performance analysis using the NASA "Eppler" code (PROFIL), commercialized for PC use as "Airfoil II". This is a simple code, but it gives excellent results, especially concerning laminar/turbulent transition, flow separation, etc.

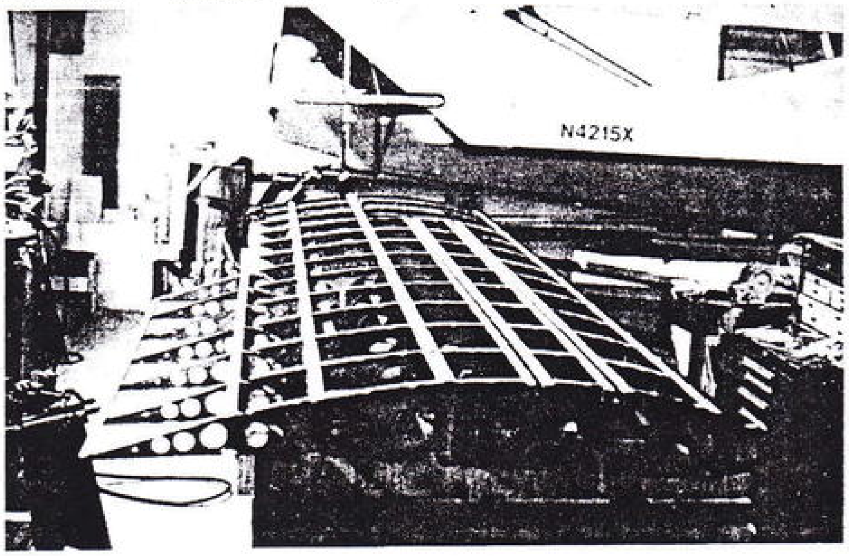
The analysis showed that the existing airfoil has a separation problem on the bottom surface near the nose at high speed, and also a similar problem on the top surface at 70 percent of the chord at all speeds. This agreed with Ken's flight test experience with the airplane, which showed a low cruise speed, high landing speed, and poor rate of climb. Accordingly, I suggested we attack the top surface problem first by installing an array of vortex generators (VT) along the entire span at .70 C, in an attempt to "fill in" the separation problem in the "low spot" there, which he did. These VGs are simple, small pieces of bent sheet metal, glued to the wing surface at a slight angle to the slipstream, and can often cure separation problems.



Comparison of the original PT-2 and new GA 30U-613.5 airfoils. Note the location of the vortex generators applied to the original wing to promote turbulent flow.

Evidently the problem was quite severe, however, since Ken reported that the VGs helped only a little bit, especially in producing slightly better aileron control, a slight improvement in rate in climb, and better handling at cruise speed. Overall, however, Ken felt that in the long run it would be better for him to invest his efforts in a new wing so in October of 1993 he tore the old wings apart, salvaged what he could and began building a new set of wings using the GA 30U-613.5 airfoil.

The new wings include a set of Hoerner wing tips that I sketched for Ken; they result in the wingspan being reduced by 16 inches. Since the new airfoil has a different lift curve that of the old wing it was necessary to decrease the angle of incidence of the new wing, and also the existing horizontal stabilizer, by 4 degrees.



Ken Rogers' new wing under construction in January of 1994.

Ken flew the airplane in August 1994 and reported these results:

	Before	After
Cruise speed	85MPH	100MPH
Top speed	90	110
Stall speed-no flaps	60-65	45
Stall speed-w/flaps and power	50-55	35
Rate of climb(FPM)	500	"better!"

In addition, Ken reports that the stall is very gentle and predictable, and that the handling characteristics are better at all speeds. An unexpected benefit is that the engine oil temperature now run 15 degrees cooler, due no doubt to the reduced drag of the new wing!

Ken is a true experimenter and is to be congratulated for his perseverance and "can do" attitude. He says, "If it ain't right, fix it". I would be glad to furnish details of the new airfoil or you can simply refer to Contact! issue #11. For details of the new wing construction I suggest you contact Ken Rogers at 1450 Konnowac Pass Road, Moxee WA 98936, or call him at (~~302~~) 248-1413. HR
509

MARCH - APRIL 1995

FIGURE 20

Aircraft Performance Prediction (see also figure 22)

Aircraft performance can be estimated from wind tunnel (computer) data using formulas from standard textbooks, which we will summarize below. There are three dimensionless coefficients of performance determined for each angle of attack as follows:

C_l = wing lift coefficient
C_d = wing drag coefficient
C_m = wing moment coefficient

From these, quantitative values of lift, drag, and pitching moment can be calculated as follows:

L = lift = $C_l \rho/2\ S_w\ V^2$, pounds
D = drag = $C_d \rho/2\ S_w\ V^2$, pounds
M = pitching moment = $C_m \rho/2\ S_w\ V^2\ C$, foot pounds

where: ρ = mass density of standard air =	.002378 @ sea level .001928 @ 7000 ft. .001756 @ 10000 ft.

S_w = wing area, square feet
V = airspeed, feet/second
C = wing MAC (average chord), feet

The term $\rho/2\ V^2$ appears in all three basic formulas, and is sometimes known as "dynamic pressure", q, pounds/ft^2.

Note that the pitching moment (in wind tunnel tests) could be measured about any reference point, but for convenience and standardization the quarter-chord point (C/4) is used. A negative pitching moment is nose down. Center of Pressure = C.P. = $C/4 - [C_m/C_l(C)]$.

The flow condition (Reynolds number = Rn) for each wind tunnel test is measured and controlled. Although Rn can be determined exactly, for practical purposes it can be approximated as follows:

Rn = 9000 (MPH) (wing chord, ft)

Thus, an airplane with a 4 ft. wing chord flying at 100 MPH has a Reynolds number of approximately 3.6×10^6 (3.6 million), sometimes written as 3.6E6. As wing chord and speed decrease, so does Rn, and performance drops slightly. Performance drops rapidly below Rn=5E5.

Test results are usually given for wings with infinite aspect ratios. No significant error is introduced down to an aspect ratio of about 7, below which the data should be adjusted in accordance with procedures found in standard text books.

Most data is taken with a perfectly smooth surface (r=0). Increasing degrees of surface roughness degrade the performance significantly. A surface factor of r=4 approximates a wing surface well contaminated with bugs. Laminar flow wings suffer most.

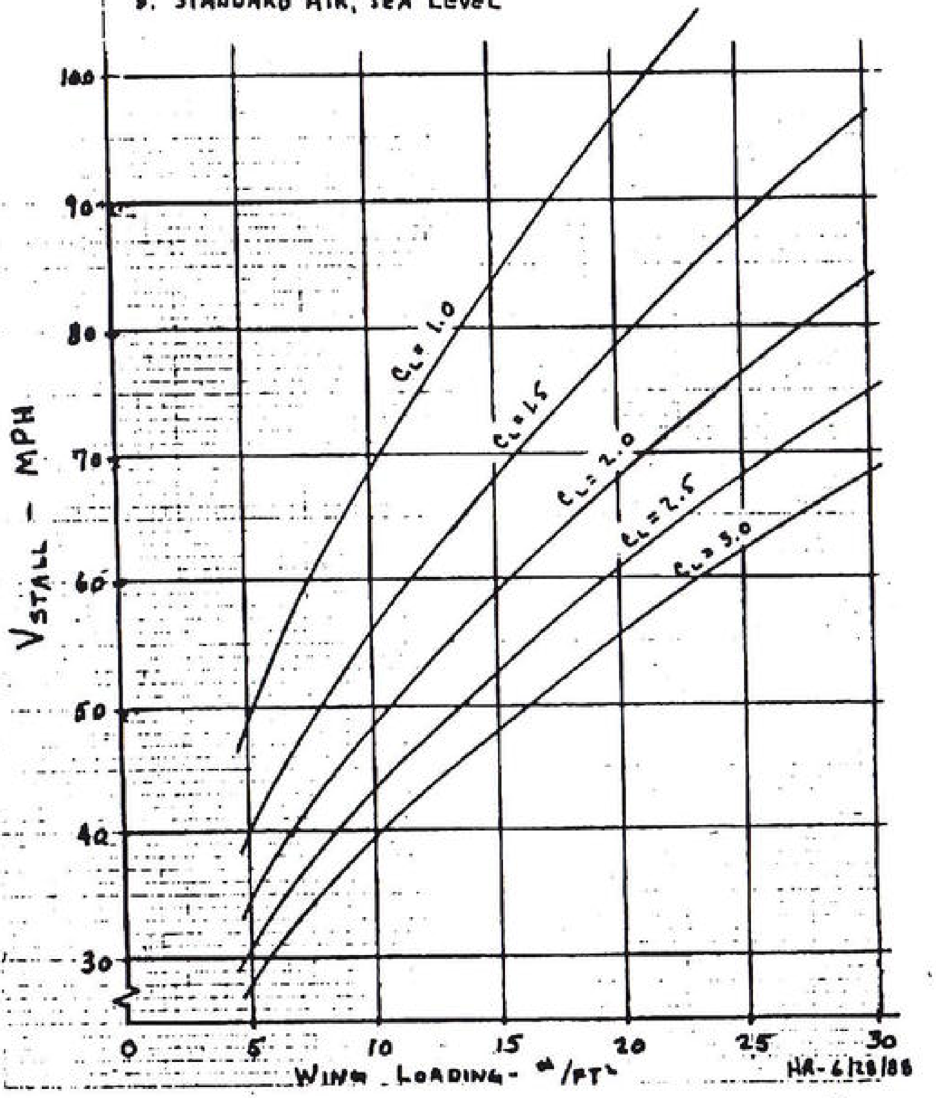
Use a "wing efficiency" factor =.85 for non-uniform span loading.

FIGURE 21

AIRCRAFT PERFORMANCE CHARTS

LANDING SPEED CHART

1. FROM $L = C_L \frac{\rho}{2} S_W V^2$
2. INCLUDES .85 WING EFFICIENCY FACTOR
3. STANDARD AIR, SEA LEVEL



CRUISING SPEED CHART

1. FROM $L = C_L \frac{\rho}{2} S_W V^2$
2. WING EFFICIENCY FACTOR = 1.0
3. STANDARD AIR, SEA LEVEL

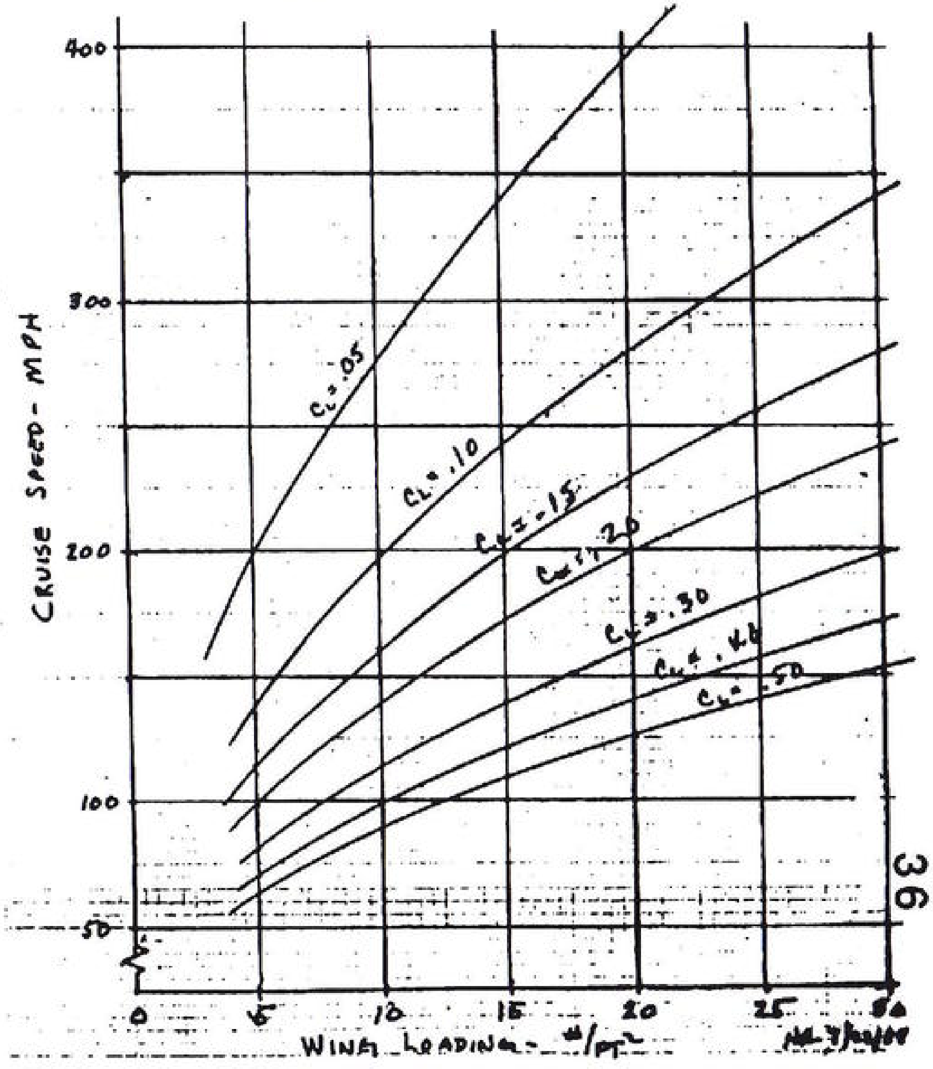
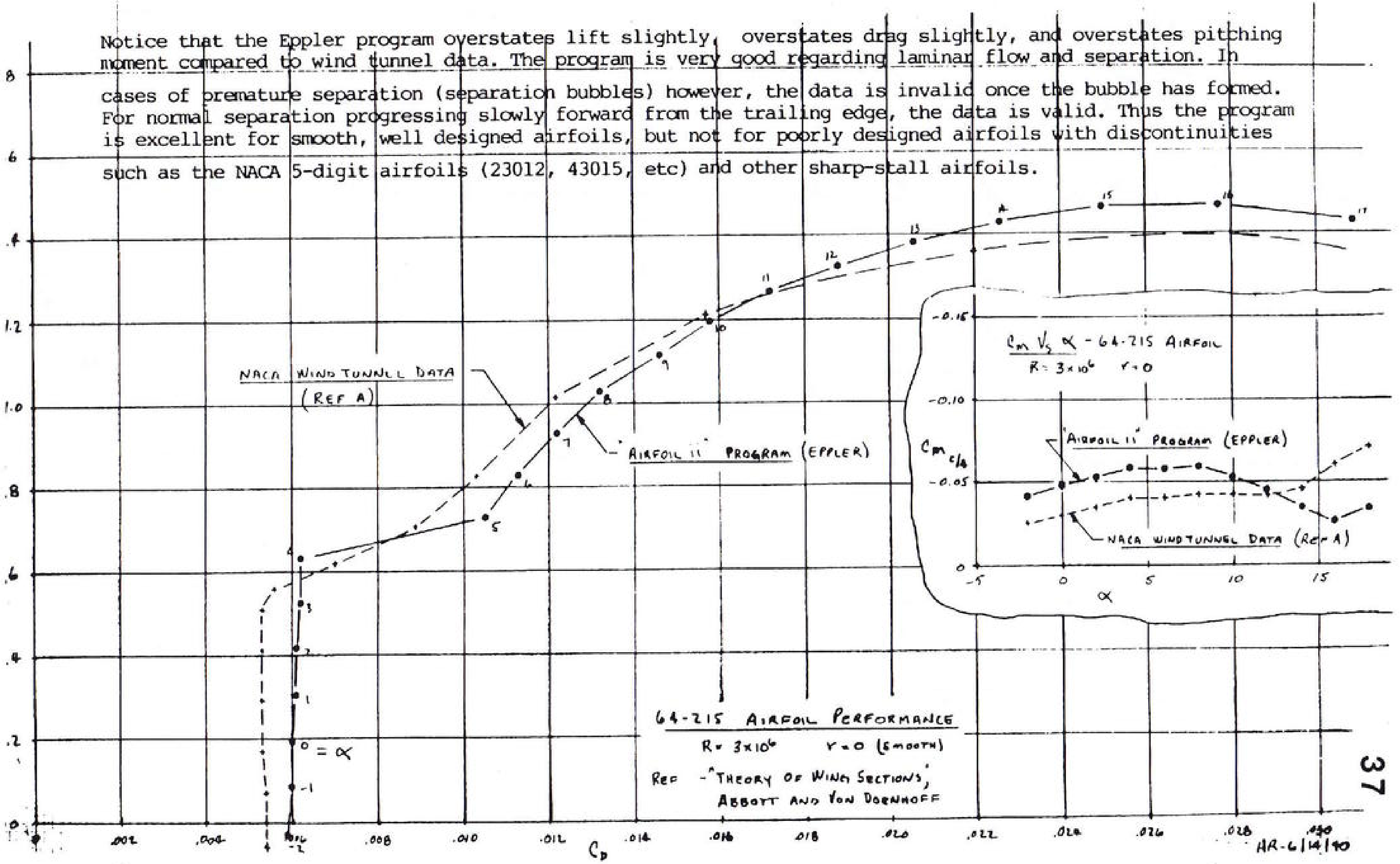


FIGURE 22

COMPUTER PROGRAM VERIFICATION

Notice that the Eppler program overstates lift slightly, overstates drag slightly, and overstates pitching moment compared to wind tunnel data. The program is very good regarding laminar flow and separation. In cases of premature separation (separation bubbles) however, the data is invalid once the bubble has formed. For normal separation progressing slowly forward from the trailing edge, the data is valid. Thus the program is excellent for smooth, well designed airfoils, but not for poorly designed airfoils with discontinuities such as the NACA 5-digit airfoils (23012, 43015, etc) and other sharp-stall airfoils.



64-215 AIRFOIL PERFORMANCE

R = 3×10⁶ r = 0 (SMOOTH)

REF - "THEORY OF WING SECTIONS", ABBOTT AND VON DOENHOFF

AR-6/14/90

TYPICAL COMPUTER ANALYSIS

(Eppler program)

```
+--------------------------------------------------+
|                    AIRFOIL-ii                    |
|                   VERSION 1.20                   |
|                                                  |
|    A program for design and analysis of airfoils |
|                                                  |
|            Portions of this program are          |
|              COPYRIGHT 1985, 1986                |
|                        by                        |
|                     AIRWARE                      |
|               All rights reserved                |
|                                                  |
|               Serial no. 86000125                |
+--------------------------------------------------+
                 Hit RETURN to continue
```

```
      37                                                                       1PANEL METHOD    A
RFOIL^a315             CA =  .28499, 7.12736   ALPHAO = 2.29 DEGREES
AIRFOIL a315             HINGE POINT AT    X/C = .0000    Y/C = .0000
 ALPHA         CM         CH      DELTA =    .00 DEGREES
  1.29     -.0529     -.09353
  2.29     -.0559     -.12757
  3.29     -.0589     -.16161
  4.29     -.0619     -.19561
  5.29     -.0650     -.22951
  6.29     -.0681     -.26328
  8.29     -.0744     -.33026
 10.29     -.0807     -.39622
 12.29     -.0871     -.46084
 14.29     -.0934     -.52381
 16.29     -.0997     -.58482
 18.29     -.1059     -.64357
 20.29     -.1120     -.69978
```

```
       37
AIRFOIL^a315                    15.00% THICKNESS          .00% FLAP        .00 DEGREES DEFLECT
ON
                                  -1.00      .00      1.00      2.00      3.00      4.00      6.00
  8.00     10.00      12.00       14.00    16.00     18.00
  N          X          Y         VELOCITY DISTRIBUTIONS FOR THE ABOVE ANGLES OF ATTACK
ELATIVE TO THE CHORD LINE
   0    1.00000     .00000      .858      .858      .857      .856      .855      .853      .849
  .845      .839       .832     .824      .815      .805
   1     .95000     .01076      .933      .936      .939      .942      .945      .947      .951
  .954      .955       .955     .955      .953      .949
   2     .90000     .02120      .975      .980      .985      .989      .994      .998     1.005
 1.011     1.015      1.019    1.021     1.022     1.021
   3     .85000     .03164     1.008     1.015     1.021     1.027     1.033     1.038     1.048
 1.057     1.064      1.070    1.075     1.078     1.081
   4     .80000     .04207     1.038     1.046     1.054     1.061     1.069     1.076     1.088
 1.100     1.110      1.119    1.126     1.132     1.136
   5     .75000     .05251     1.070     1.080     1.089     1.099     1.107     1.116     1.131
 1.146     1.159      1.170    1.180     1.189     1.196
   6     .70000     .06285     1.111     1.122     1.133     1.144     1.154     1.165     1.184
 1.201     1.217      1.232    1.245     1.257     1.267
   7     .65000     .07258     1.149     1.163     1.176     1.188     1.201     1.213     1.233
 1.257     1.276      1.295    1.311     1.326     1.340
```

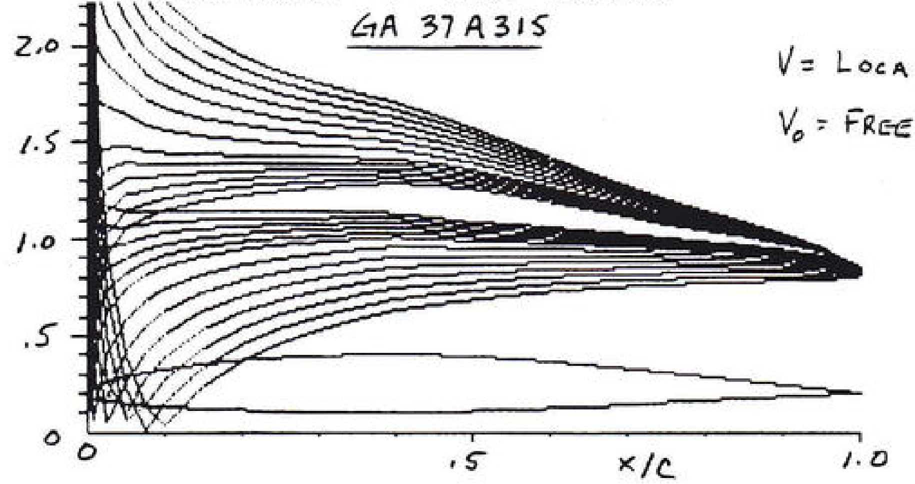
FIGURE 24-1

```
  8    .60000    .08134   1.186   1.202   1.217   1.231   1.246   1.260   1.287
1.312   1.335    1.357    1.378   1.396   1.413
  9    .55000    .08885   1.221   1.239   1.256   1.273   1.290   1.306   1.338
1.367   1.395    1.421    1.446   1.469   1.490
 10    .50000    .09477   1.256   1.276   1.296   1.315   1.335   1.353   1.390
1.424   1.457    1.489    1.518   1.545   1.571
 11    .45000    .09856   1.274   1.297   1.320   1.342   1.364   1.385   1.427
1.467   1.505    1.541    1.576   1.608   1.639
 12    .40000    .10011   1.285   1.311   1.336   1.361   1.386   1.411   1.458
1.504   1.548    1.590    1.630   1.668   1.704
 13    .35000    .09920   1.275   1.304   1.333   1.361   1.389   1.417   1.470
1.522   1.572    1.620    1.666   1.710   1.752
 14    .30000    .09610   1.260   1.292   1.324   1.356   1.387   1.418   1.478
1.537   1.594    1.649    1.701   1.752   1.801
 15    .25000    .09099   1.241   1.278   1.314   1.350   1.385   1.420   1.489
1.556   1.621    1.684    1.745   1.804   1.860
 16    .20000    .08381   1.221   1.263   1.304   1.346   1.387   1.427   1.506
1.584   1.660    1.733    1.805   1.874   1.942
 17    .15000    .07415   1.190   1.239   1.288   1.337   1.385   1.433   1.527
1.619   1.709    1.798    1.884   1.967   2.049
 18    .10000    .06134   1.143   1.204   1.264   1.324   1.384   1.443   1.560
1.675   1.788    1.899    2.007   2.113   2.217
 19    .07500    .05331   1.107   1.177   1.246   1.315   1.384   1.452   1.587
1.720   1.851    1.980    2.106   2.230   2.351
 20    .05000    .04360   1.059   1.143   1.227   1.311   1.395   1.478   1.643
1.806   1.966    2.124    2.280   2.433   2.583
 21    .02500    .03036    .917   1.028   1.139   1.249   1.359   1.469   1.687
1.903   2.116    2.327    2.535   2.740   2.942
 22    .01250    .02105    .745    .885   1.024   1.162   1.300   1.438   1.712
1.985   2.254    2.521    2.785   3.046   3.303
 23    .00750    .01618    .613    .774    .935   1.095   1.255   1.414   1.732
2.047   2.360    2.670    2.977   3.280   3.579
 24    .00500    .01318    .507    .685    .863   1.040   1.217   1.394   1.747
2.097   2.445    2.789    3.131   3.468   3.801
 25    .00250    .00920    .307    .503    .700    .896   1.092   1.288   1.678
2.066   2.451    2.834    3.213   3.588   3.959
 26    .00000    .00000    .376    .134    .109    .352    .595    .837   1.321
1.804   2.284    2.762    3.236   3.706   4.172
 27    .00250   -.00790   1.061    .828    .593    .359    .125    .110    .579
1.047   1.514    1.979    2.442   2.901   3.357
 28    .00500   -.01068   1.206    .997    .788    .579    .369    .159    .260
 .679   1.098    1.515    1.930   2.343   2.753
 29    .00750   -.01254   1.224   1.041    .857    .674    .490    .306    .062
 .431    .798    1.165    1.530   1.893   2.254
 30    .01250   -.01525   1.215   1.065    .915    .764    .613    .462    .160
 .143    .445     .747    1.048   1.348   1.646
 31    .02500   -.01980   1.173   1.063    .954    .843    .733    .622    .400
 .178    .045     .267     .490    .711    .932
 32    .05000   -.02594   1.138   1.060    .981    .901    .822    .742    .582
 .421    .260     .098     .064    .225    .387
 33    .07500   -.03073   1.139   1.074   1.009    .943    .877    .811    .678
 .544    .410     .275     .140    .004    .132
 34    .10000   -.03464   1.141   1.084   1.028    .971    .914    .856    .740
 .623    .506     .388     .269    .150    .031
 35    .15000   -.04049   1.136   1.091   1.045   1.000    .954    .907    .814
 .719    .623     .527     .430    .333    .235
 36    .20000   -.04465   1.135   1.096   1.058   1.019    .980    .940    .861
 .780    .698     .615     .532    .448    .364
 37    .25000   -.04753   1.133   1.100   1.067   1.033    .999    .964    .894
 .823    .751     .679     .605    .531    .455
 38    .30000   -.04930   1.132   1.102   1.073   1.043   1.012    .982    .920
 .856    .792     .727     .661    .594    .526
 39    .35000   -.05006   1.132   1.106   1.079   1.052   1.025    .998    .942
 .885    .826     .767     .707    .646    .584

 40    .40000   -.04963   1.130   1.107   1.083   1.059   1.034   1.009    .958
 .907    .854     .800     .745    .689    .632
 41    .45000   -.04770   1.109   1.088   1.067   1.046   1.024   1.002    .956
 .910    .862     .814     .764    .714    .662
 42    .50000   -.04479   1.085   1.066   1.047   1.028   1.009    .989    .948
 .907    .864     .820     .775    .730    .683
 43    .55000   -.04149   1.070   1.053   1.036   1.019   1.001    .983    .947
 .909    .870     .830     .789    .748    .705
 44    .60000   -.03778   1.055   1.040   1.025   1.009    .994    .978    .944
 .910    .875     .839     .801    .763    .724
 45    .65000   -.03364   1.038   1.025   1.012    .998    .984    .969    .939
 .909    .877     .843     .809    .774    .738
 46    .70000   -.02915   1.020   1.009    .997    .984    .972    .959    .932
 .905    .876     .846     .815    .783    .750
 47    .75000   -.02443   1.001    .991    .981    .970    .959    .947    .924
 .899    .873     .846     .818    .789    .759
 48    .80000   -.01961    .983    .974    .965    .955    .946    .936    .915
 .893    .870     .846     .820    .794    .767
 49    .85000   -.01478    .963    .956    .948    .940    .932    .923    .905
 .886    .866     .844     .822    .799    .774
 50    .90000   -.00996    .941    .934    .928    .921    .915    .907    .892
 .876    .859     .840     .821    .800    .779
 51    .95000   -.00514    .909    .905    .900    .895    .890    .884    .873
 .860    .846     .831     .815    .798    .780
```

FIGURE 14-2

```
AIRFOIL 37a315               15.00% THICKNESS           .00% FLAP       .00 DEGREES DEFLEC
ON
                              -1.00       .00       1.00       2.00       3.00       4.00      6.0
  8.00      10.00     12.00    14.00     16.00     18.00
   N          X         Y     VELOCITY DISTRIBUTIONS FOR THE ABOVE ANGLES OF ATTACK
ELATIVE TO THE CHORD LINE
  52    1.00000    .00000      .858      .858      .857      .856      .855      .853      .84
  .845      .839      .832     .824      .815      .805
```

Airfoil number 8 Date - 2-26-1994



GA 37A315

V = LOCAL VELOCITY

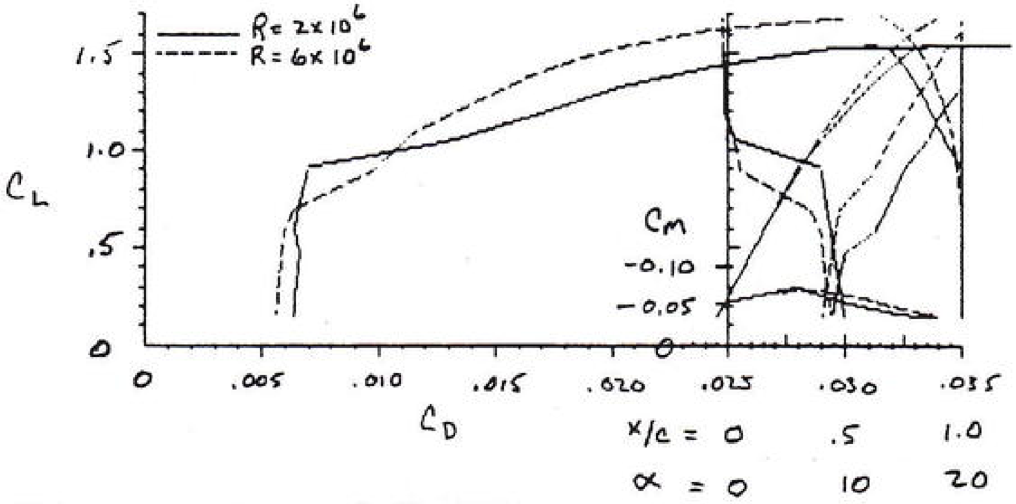
V_0 = FREE STREAM VELOCITY

```
SUMMARY     AIRFOIL 37a315              ANGLE OF ATTACK RELATIVE TO THE CHORD LINE
   ALPHAO = 2.29 DEGREES
 * INDICATES BUBBLE ANALOG LONGER THAN .030

            R =   2000000    MU = 3   |    R =   6000000    MU = 3
                                      |
 ALPHA = -1.00 DEGREES                |
        1 S TURB    S SEP      CD     |  2 S TURB    S SEP      CD
  UPPER    .5045    .0000    .0034    |     .5573    .0000    .0031
  LOWER    .5565    .0000    .0029    |     .5948    .0000    .0025
 TOTAL     CL =    .142 CD = .0064    |    CL =    .142 CD = .0056
               CM = -.0529            |        CM = -.0529
                                      |
 ALPHA =    .00 DEGREES               |
        1 S TURB    S SEP      CD     |  2 S TURB    S SEP      CD
  UPPER    .5178    .0000    .0037    |     .5733    .0000    .0033
  LOWER    .5410    .0000    .0027    |     .5845    .0000    .0024
 TOTAL     CL =    .252 CD = .0064    |    CL =    .252 CD = .0057
               CM = -.0559            |        CM = -.0559
                                      |
 ALPHA =   1.00 DEGREES               |
        1 S TURB    S SEP      CD     |  2 S TURB    S SEP      CD
  UPPER    .5349    .0000    .0040    |     .5858    .0000    .0035
  LOWER    .5240    .0000    .0025    |     .5723    .0000    .0022
 TOTAL     CL =    .362 CD = .0065    |    CL =    .362 CD = .0057
               CM = -.0589            |        CM = -.0589
                                      |
 ALPHA =   2.00 DEGREES               |
        1 S TURB    S SEP      CD     |  2 S TURB    S SEP      CD
  UPPER    .5493    .0000    .0043    |     .5971    .0000    .0038
  LOWER    .5027    .0000    .0023    |     .5570    .0000    .0021
 TOTAL     CL =    .472 CD = .0066    |    CL =    .472 CD = .0058
               CM = -.0619            |        CM = -.0619
                                      |
 ALPHA =   3.00 DEGREES               |
        1 S TURB    S SEP      CD     |  2 S TURB    S SEP      CD
  UPPER    .5630    .0000    .0046    |     .6067    .0000    .0040
  LOWER    .3738    .0000    .0018    |     .5428    .0000    .0019
 TOTAL     CL =    .582 CD = .0064    |    CL =    .582 CD = .0059
               CM = -.0650            |        CM = -.0650
                                      |
 ALPHA =   4.00 DEGREES               |
        1 S TURB    S SEP      CD     |  2 S TURB    S SEP      CD
  UPPER    .5794    .0000    .0049    |     .6491    .0000    .0045
  LOWER    .3277    .0000    .0016    |     .5272    .0000    .0018
 TOTAL     CL =    .692 CD = .0065    |    CL =    .692 CD = .0064
               CM = -.0681            |        CM = -.0681
                                      |
 ALPHA =   6.00 DEGREES               |
        1 S TURB    S SEP      CD     |  2 S TURB    S SEP      CD
  UPPER    .6071    .0000    .0057    |     .9476    .0196    .0084
  LOWER    .2435    .0000    .0012    |     .3756    .0000    .0013
 TOTAL     CL =    .912 CD = .0070    |    CL =    .892 CD = .0097
               CM = -.0744            |        CM = -.0696
```

FIGURE 24-3

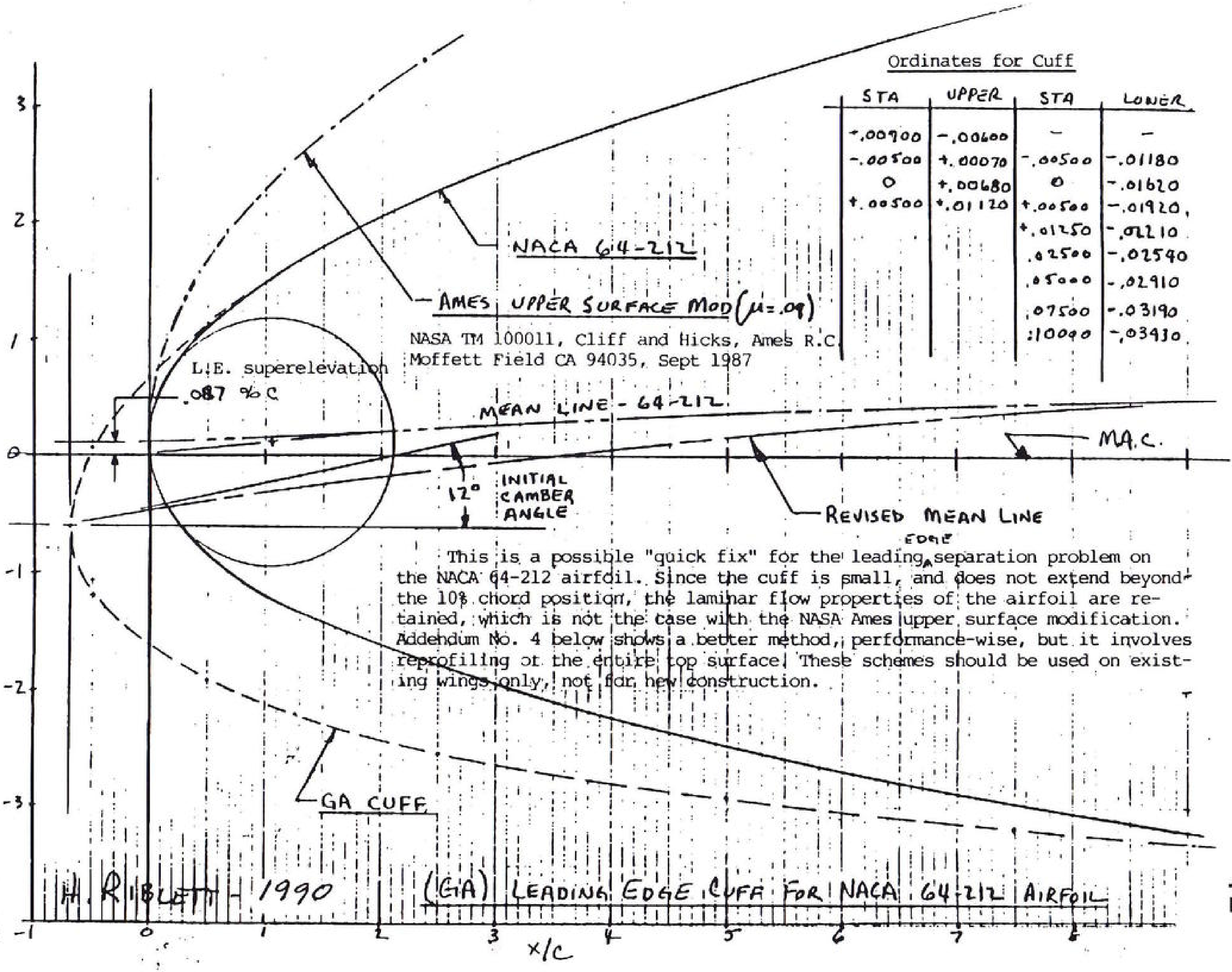
```
SUMMARY    AIRFOIL a315                ANGLE OF ATTACK RELATIVE TO THE CHORD LINE
    ALPHAO = 2.29 DEGREES
 * INDICATES BUBBLE ANALOG LONGER THAN .030

             R =   2000000     MU = 3  |  R =   6000000     MU = 3

ALPHA =   8.00 DEGREES                 |
        1 S TURB     S SEP      CD     | 2 S TURB     S SEP      CD
 UPPER     .9676     .0711    .0124    |    .9812     .0419    .0105
 LOWER     .1444     .0000    .0009    |    .2961     .0000    .0010
TOTAL     CL = 1.053 CD =     .0133    |  CL = 1.086 CD =     .0115
               CM = -.0631             |       CM = -.0699

ALPHA =  10.00 DEGREES                 |
        1 S TURB     S SEP      CD     | 2 S TURB     S SEP      CD
 UPPER    1.0167     .1303    .0164    |   1.0167     .0822    .0136
 LOWER     .0881     .0000    .0007    |    .2081     .0000    .0007
TOTAL     CL = 1.194 CD =     .0171    |  CL = 1.252 CD =     .0143
               CM = -.0550             |       CM = -.0651

ALPHA =  12.00 DEGREES                 |
        1 S TURB     S SEP      CD     | 2 S TURB     S SEP      CD
 UPPER    1.0186     .1871    .0199    |   1.0186     .1294    .0163
 LOWER     .0000     .0000    .0005    |    .1196     .0000    .0005
TOTAL     CL = 1.324 CD =     .0203    |  CL = 1.400 CD =     .0168
               CM = -.0480             |       CM = -.0586

ALPHA =  14.00 DEGREES                 |
        1 S TURB     S SEP      CD     | 2 S TURB     S SEP      CD
 UPPER    1.0197     .2483    .0241    |   1.0197     .1880    .0197
 LOWER     .0000     .0000    .0004    |    .0690     .0000    .0004
TOTAL     CL = 1.436 CD =     .0245    |  CL = 1.522 CD =     .0201
               CM = -.0416             |       CM = -.0503

ALPHA =  16.00 DEGREES                 |
        1 S TURB     S SEP      CD     | 2 S TURB     S SEP      CD
 UPPER    1.0205     .3117    .0293    |   1.0210     .2545    .0239
 LOWER     .0000     .0000    .0004    |    .0000     .0000    .0002
TOTAL     CL = 1.530 CD =     .0296    |  CL = 1.619 CD =     .0241
               CM = -.0372             |       CM = -.0426

ALPHA =  18.00 DEGREES                 |
        1 S TURB     S SEP      CD     | 2 S TURB     S SEP      CD
 UPPER    1.0230     .4174    .0368    |   1.0235     .3322    .0297
 LOWER     .0000     .0000    .0003    |    .0000     .0000    .0002
TOTAL     CL = 1.541 CD =     .0372    |  CL = 1.682 CD =     .0299
               CM = -.0352             |       CM = -.0370
                    Airfoil number    0    Date -  2-26-1994
```

GA 37 A315



```
This run made on   2-26-1994

             At    1:38:48.77


   Run time was [redacted]
```

FIGURE 24-4


3
2
1
0
-1
-2
-3
-1
0
1
2
3
4
5
6
7
8
x/c
NACA 64-212
AMES UPPER SURFACE MOD (μ=.09)
L.E. superelevation
.087 %C
MEAN LINE - 64-212
M.A.C.
12°
INITIAL CAMBER ANGLE
REVISED MEAN LINE
GA CUFF

Ordinates for Cuff

STA	UPPER	STA	LOWER
+.00900	-.00600	–	–
-.00500	+.00070	-.00500	-.01180
0	+.00680	0	-.01620
+.00500	+.01120	+.00500	-.01920
		+.01250	-.02210
		.02500	-.02590
		.05000	-.02910
		.07500	-.03190
		.10090	-.03430

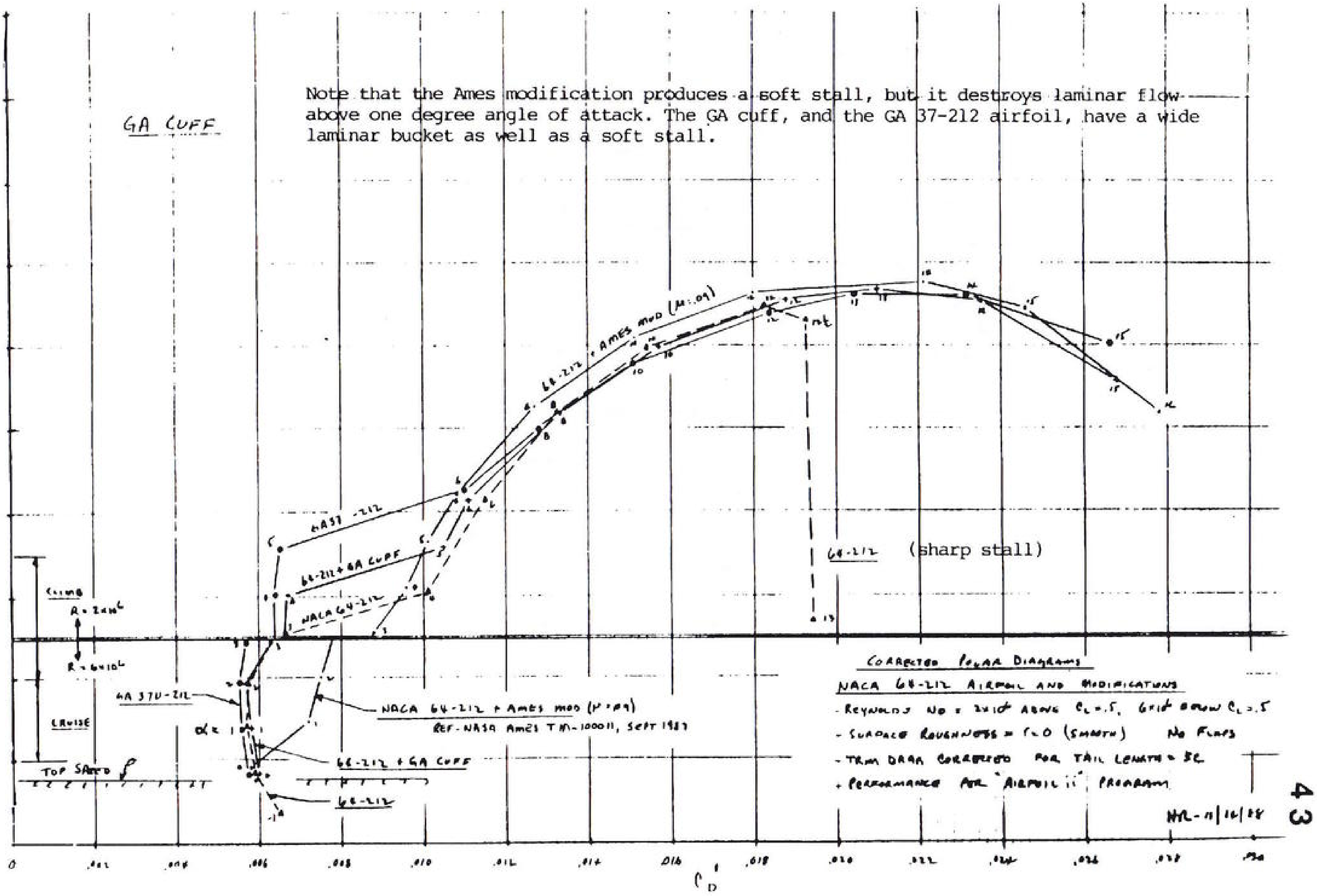
NASA TM 100011, Cliff and Hicks, Ames R.C.
Moffett Field CA 94035, Sept 1987

This is a possible "quick fix" for the leading edge separation problem on the NACA 64-212 airfoil. Since the cuff is small, and does not extend beyond the 10% chord position, the laminar flow properties of the airfoil are retained, which is not the case with the NASA Ames upper surface modification. Addendum No. 4 below shows a better method, performance-wise, but it involves reprofiling of the entire top surface. These schemes should be used on existing wings only, not for new construction.

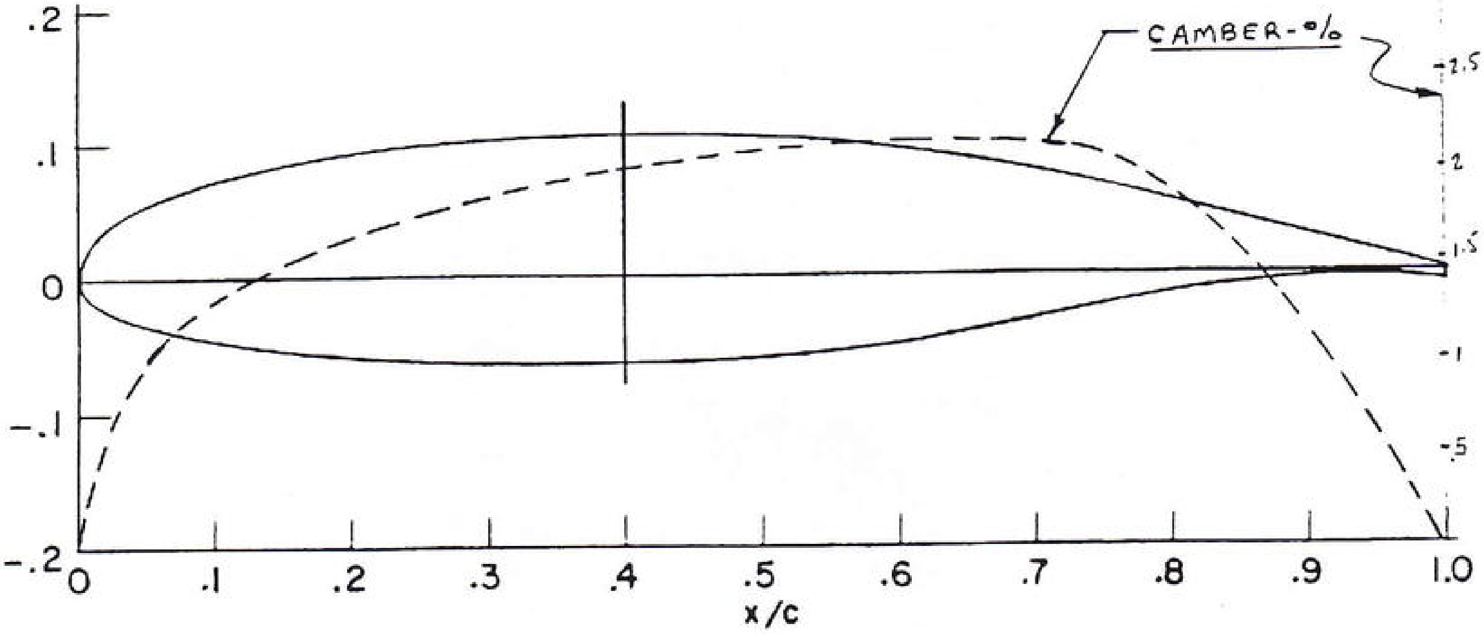
H. RIBLETT - 1990 (GA) LEADING EDGE CUFF FOR NACA 64-212 AIRFOIL

FIGURE 25

Note that the Ames modification produces a soft stall, but it destroys laminar flow above one degree angle of attack. The GA cuff, and the GA 37-212 airfoil, have a wide laminar bucket as well as a soft stall.



NASA GA (W)-1 AIRFOIL [LS(1)-0417]



1974

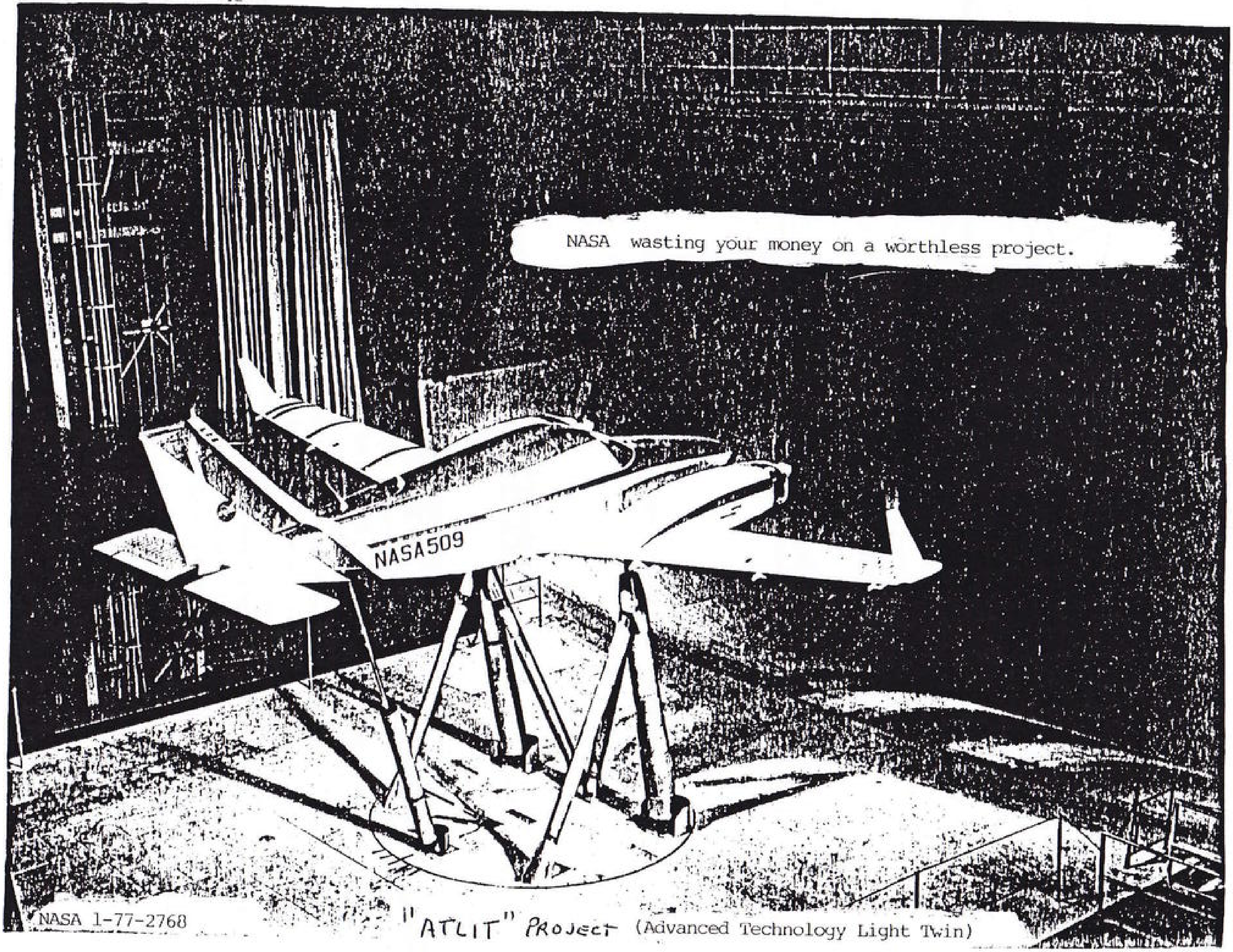
Figure 1.- Section shape for NASA GA(W)-1 airfoil. (17% THICK)

gust 5, 1974 **BUSINESS AVIATION** Page 44

FIRST FLIGHT OF SENECA WITH MODIFIED WING SCHEDULED – National Aeronautics and Space Administration will supervise the first flight of a Piper Seneca next month modified with a new wing design that promises major performance improvements. The wing assembly for the aircraft – designated as the NASA GAW-1 – is being built and installed on the aircraft at Piper's Lakeland, Fla., facility. The low-speed airfoil features full-span power flaps and spoilers for lateral control. Wind tunnel tests with models have indicated that the new airfoil will provide increases of 90 per cent in range, 80 per cent in the single-engine-rate-of-climb, 30 per cent in maximum lift, 50 per cent in L/D and 10 per cent in cruise speed. Although the wing is being built by Piper and installed on a Seneca, NASA spokesmen emphasized that the program is designed to provide a more efficient wing for a wide range of general aviation aircraft, not just the Seneca. A full flight test program will be conducted at NASA's Langley, Va., research facility after initial flights are completed in Florida.

"ATLIT" PROJECT

FIGURE 27



NASA wasting your money on a worthless project.

NASA 1-77-2768

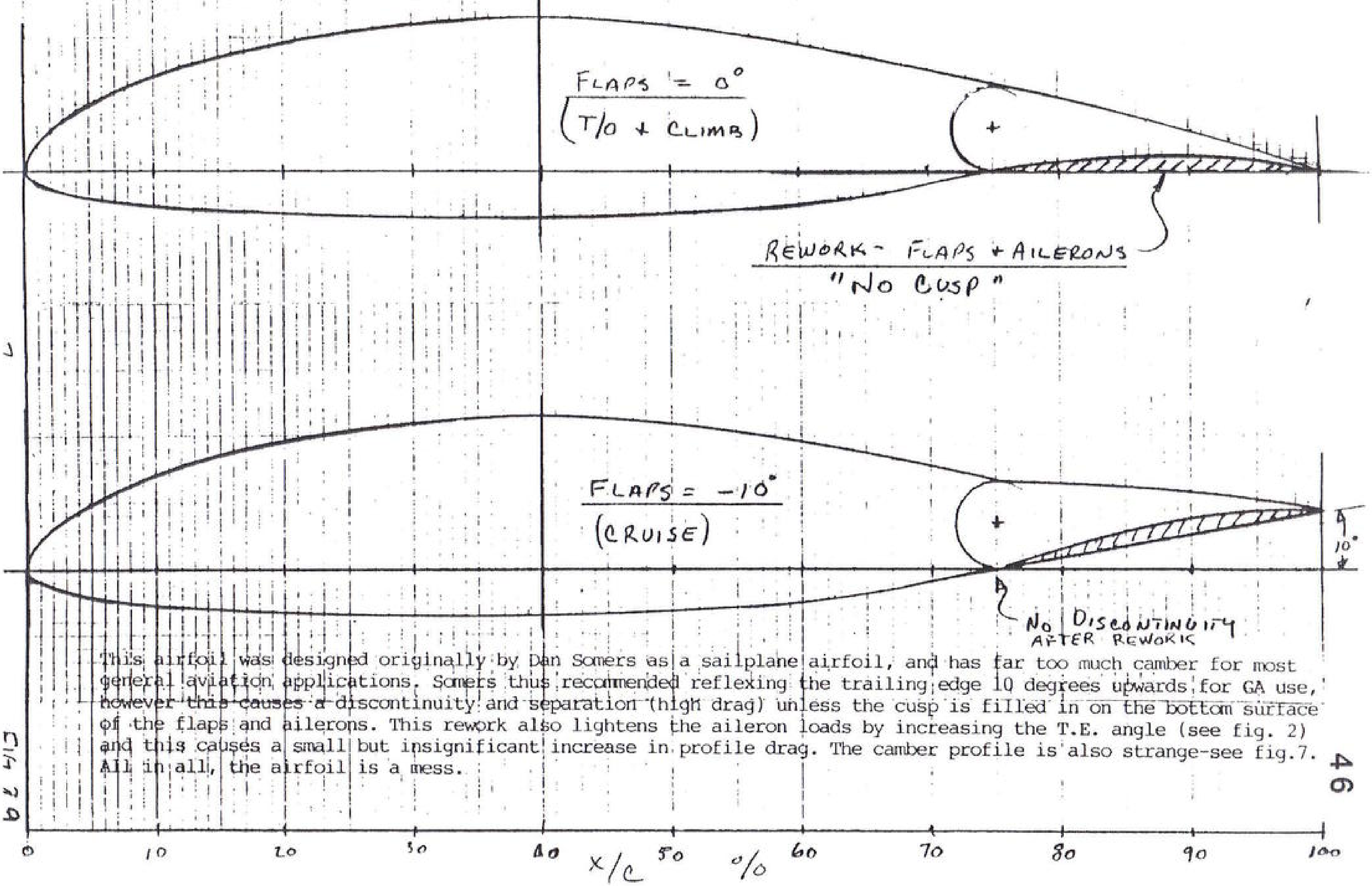
"ATLIT" PROJECT (Advanced Technology Light Twin)

FIG 28

NASA WLF(1)-0215(F) AIRFOIL

SEE NASA TP# 1865, JUNE 1981

Used on Prescott Pusher, Poliwagen, Wheeler Express,
Lancair (extensive rework), and others.


FLAPS = 0°
(T/O + CLIMB)
REWORK - FLAPS + AILERONS
"NO CUSP"
FLAPS = -10°
(CRUISE)
10°
NO DISCONTINUITY
AFTER REWORK
0
10
20
30
40
50
60
70
80
90
100
X/C %

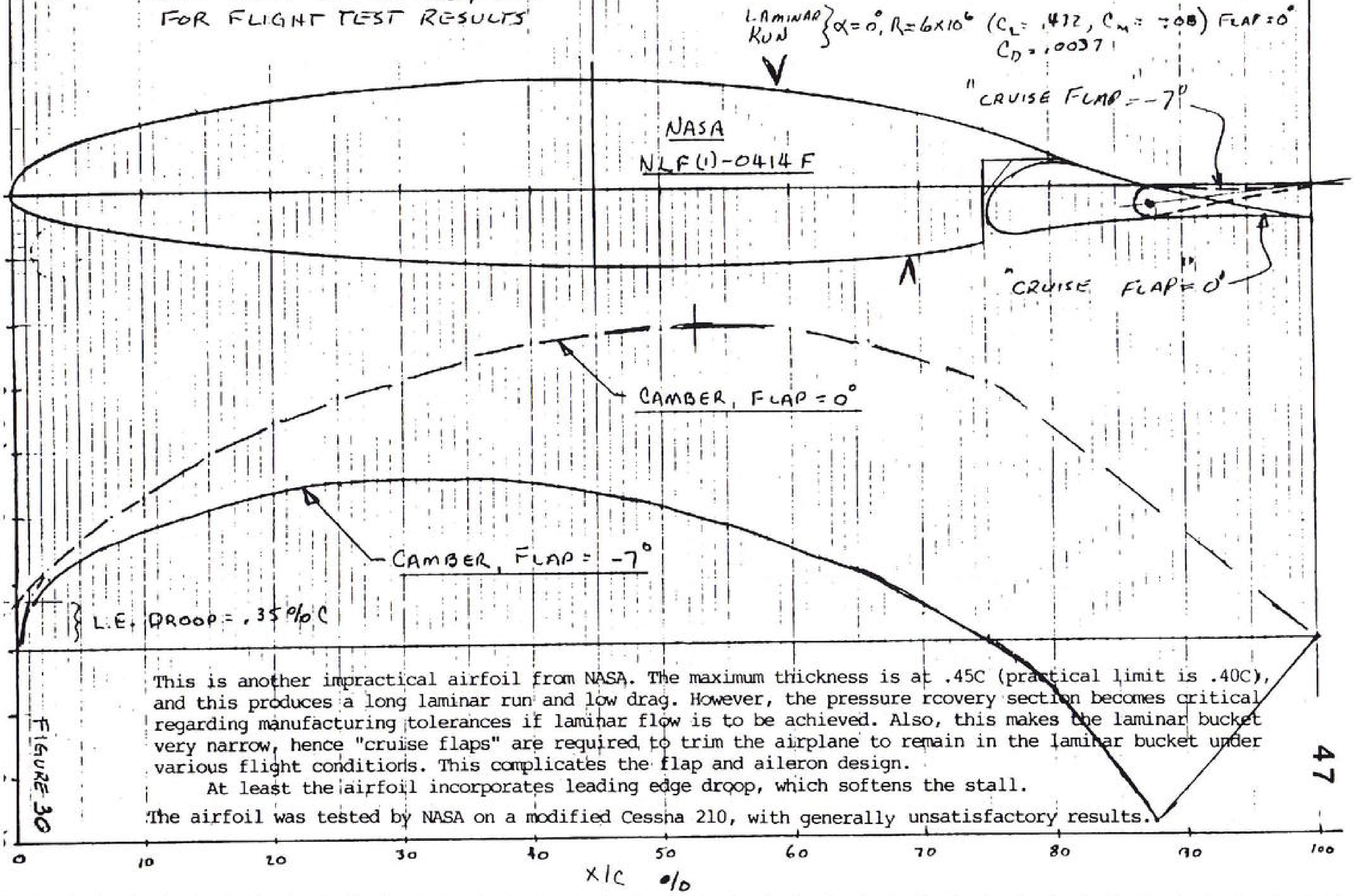
This airfoil was designed originally by Dan Somers as a sailplane airfoil, and has far too much camber for most general aviation applications. Somers thus recommended reflexing the trailing edge 10 degrees upwards for GA use, however this causes a discontinuity and separation (high drag) unless the cusp is filled in on the bottom surface of the flaps and ailerons. This rework also lightens the aileron loads by increasing the T.E. angle (see fig. 2) and this causes a small but insignificant increase in profile drag. The camber profile is also strange-see fig.7. All in all, the airfoil is a mess.

FIG 29

NASA NLF(1)-0414(F) AIRFOIL

H. RIBLETT 8/11/88

SEE NASA TM 85788, 1984
FOR FLIGHT TEST RESULTS


LAMINAR RUN
$\alpha = 0°, R = 6 \times 10^6$ (C_L = .472, C_M = -.08) FLAP = 0°
C_D = .0037
NASA
NLF(1)-0414 F
"CRUISE FLAP" = -7°
"CRUISE FLAP" = 0°
CAMBER, FLAP = 0°
CAMBER, FLAP = -7°
L.E. DROOP = .35 % C
0
10
20
30
40
50
60
70
80
90
100
X/C %

This is another impractical airfoil from NASA. The maximum thickness is at .45C (practical limit is .40C), and this produces a long laminar run and low drag. However, the pressure rcovery section becomes critical regarding manufacturing tolerances if laminar flow is to be achieved. Also, this makes the laminar bucket very narrow, hence "cruise flaps" are required to trim the airplane to remain in the laminar bucket under various flight conditions. This complicates the flap and aileron design.

At least the airfoil incorporates leading edge droop, which softens the stall.

The airfoil was tested by NASA on a modified Cessna 210, with generally unsatisfactory results.

FIGURE 30

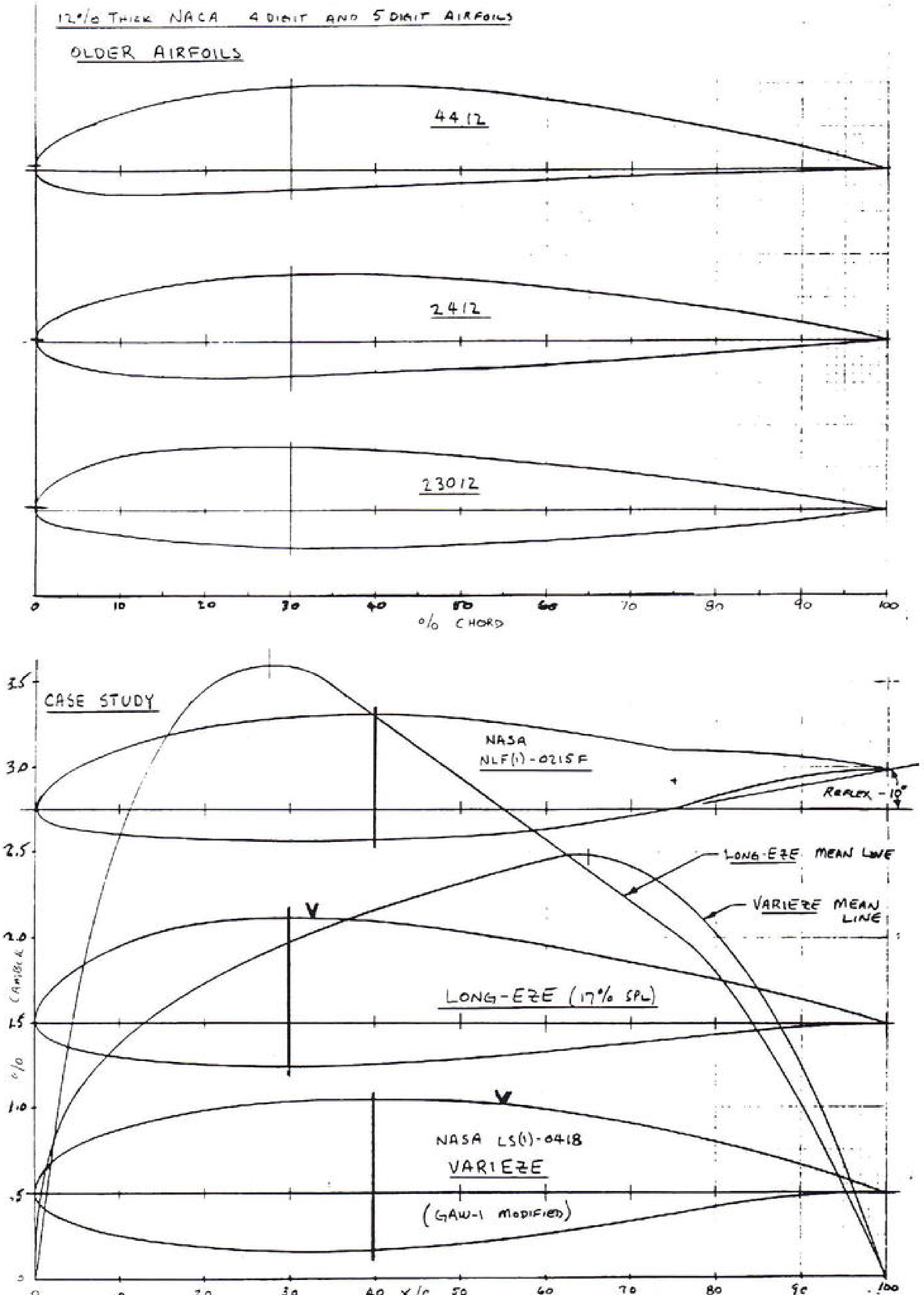
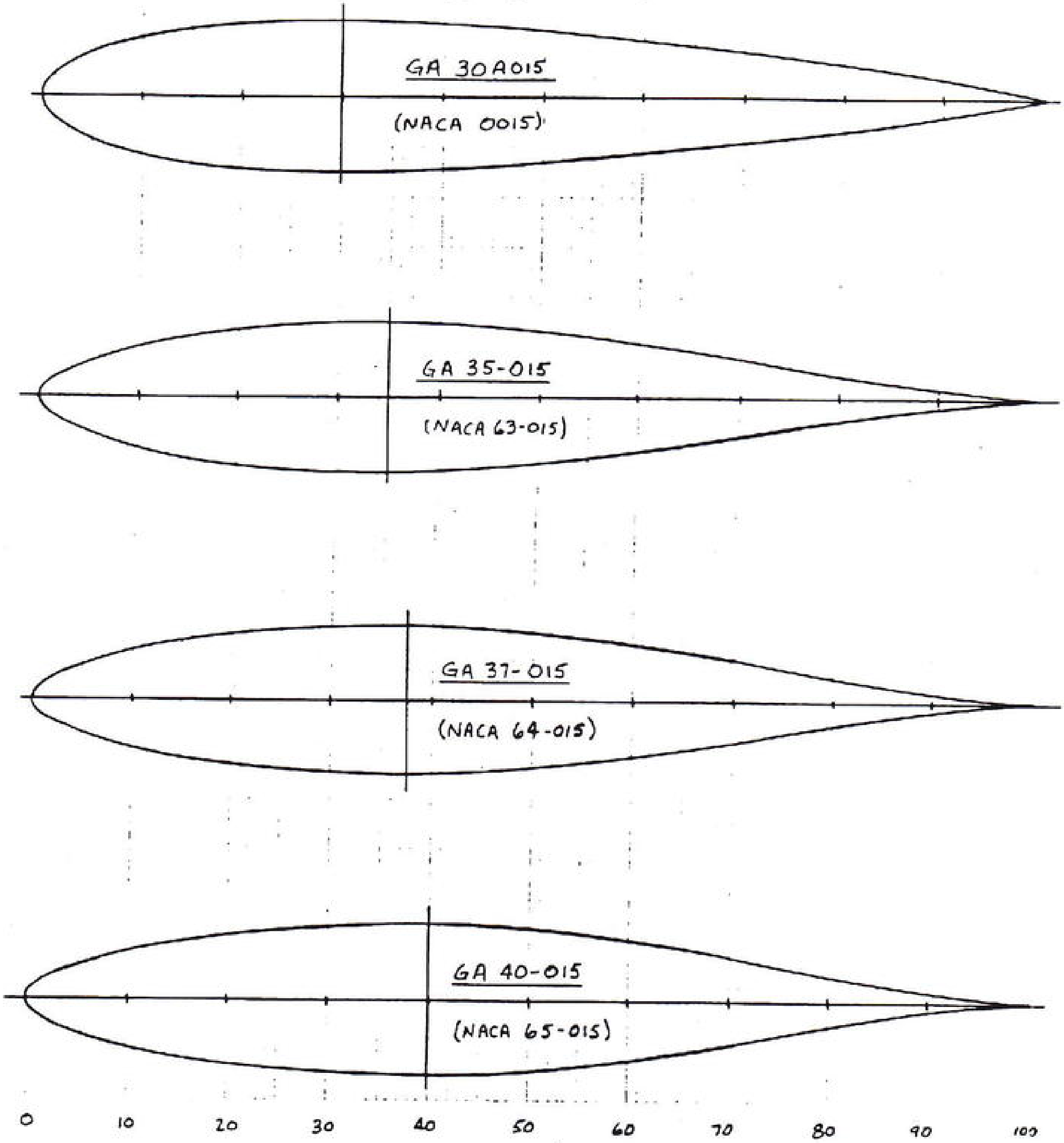

12% THICK NACA 4 DIGIT AND 5 DIGIT AIRFOILS
OLDER AIRFOILS
4412
2412
23012
0
10
20
30
40
50
60
70
80
90
100
% CHORD
CASE STUDY
NASA
NLF(1)-0215F
REFLEX -10°
LONG-EZE MEAN LINE
VARIEZE MEAN LINE
V
LONG-EZE (17% SPL)
NASA LS(1)-0418
VARIEZE
(GAW-1 MODIFIED)
3.5
3.0
2.5
2.0
1.5
1.0
.5
0
% CAMBER
X/C

FIG 31

APPENDIX I - Basic Thickness Forms.

I - BASIC THICKNESS FORMS (Symmetrical shapes, 15% thick)

A. "CUSP" TYPE THICKNESS DISTRIBUTIONS.


GA 30A015
(NACA 0015)
GA 35-015
(NACA 63-015)
GA 37-015
(NACA 64-015)
GA 40-015
(NACA 65-015)
0
10
20
30
40
50
60
70
80
90
100

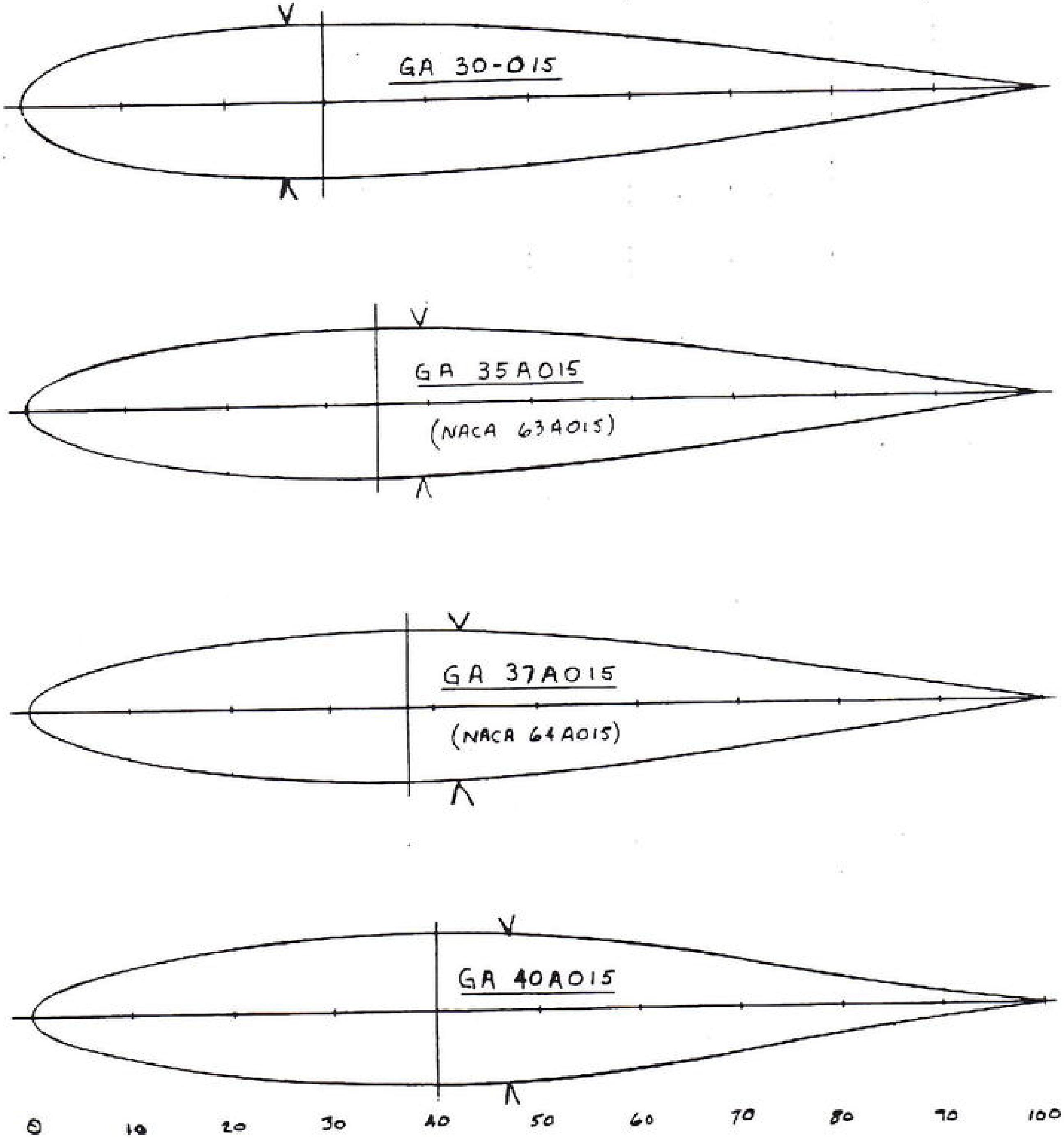
Note- For thicknesses other than 15%, these shapes may be scaled directly up or down to the desired thickness.

FIGURE I-1

I - BASIC THICKNESS FORMS (Symmetrical shapes, 15% thick)

B. PREFERRED 'NO CUSP' THICKNESS DISTRIBUTIONS

V = TRANSITION, $R = 6 \times 10^6$


V
GA 30-015
Λ
V
GA 35A015
(NACA 63A015)
Λ
V
GA 37A015
(NACA 64A015)
Λ
V
GA 40A015
Λ
0
10
20
30
40
50
60
70
80
90
100

Note- For thicknesses other than 15%, these shapes may be scaled directly up or down to the desired thickness.

FIG I-2

APPENDIX IA - Cusp Type Thickness Distributions (GA 30A, 35-, 37-, 40-)

Feb 94	0015	63-015	64-015	65-015
Sta	Ord	Ord	Ord	Ord
0.00	0.000	0.000	0.000	0.000
0.25	1.090	0.875	0.865	0.830
0.50	1.527	1.204	1.208	1.124
0.75	1.856	1.462	1.456	1.356
1.25	2.367	1.878	1.842	1.702
2.50	3.268	2.610	2.528	2.324
5.00	4.443	3.648	3.504	3.245
7.50	5.250	4.427	4.240	3.959
10.00	5.853	5.055	4.842	4.555
15.00	6.681	6.011	5.785	5.504
20.00	7.172	6.693	6.480	6.223
25.00	7.427	7.155	6.985	6.764
30.00	7.502	7.421	7.319	7.152
35.00	7.436	7.500	7.482	7.396
40.00	7.254	7.386	7.473	7.498
45.00	6.976	7.099	7.224	7.427
50.00	6.618	6.665	6.810	7.168
55.00	6.191	6.108	6.266	6.720
60.00	5.704	5.453	5.620	6.118
65.00	5.166	4.721	4.895	5.403
70.00	4.580	3.934	4.113	4.600
75.00	3.950	3.119	3.296	3.744
80.00	3.279	2.310	2.472	2.858
85.00	2.566	1.541	1.677	1.977
90.00	1.810	0.852	0.950	1.144
95.00	1.008	0.300	0.346	0.428
100.00	0.158	0.000	0.000	0.000

APPENDIX IB - No Cusp Thickness Distributions (GA 30-, 35A, 37A, 40A)

Feb 94	30-015	63A015	64A015	40A015
Sta	Ord	Ord	Ord	Ord
0.00	0.000	0.000	0.000	0.000
0.25	1.090	0.875	0.855	0.830
0.50	1.527	1.203	1.193	1.124
0.75	1.856	1.448	1.436	1.356
1.25	2.367	1.844	1.815	1.702
2.50	3.268	2.579	2.508	2.324
5.00	4.443	3.618	3.477	3.245
7.50	5.250	4.382	4.202	3.959
10.00	5.853	4.997	4.799	4.555
15.00	6.681	5.942	5.732	5.504
20.00	7.172	6.619	6.423	6.223
25.00	7.427	7.091	6.926	6.764
30.00	7.502	7.384	7.270	7.152
35.00	7.440	7.496	7.463	7.396
40.00	7.265	7.435	7.487	7.498
45.00	7.000	7.215	7.313	7.427
50.00	6.655	6.858	6.978	7.168
55.00	6.240	6.387	6.517	6.720
60.00	5.755	5.820	5.956	6.118
65.00	5.190	5.173	5.311	5.403
70.00	4.540	4.468	4.600	4.600
75.00	3.817	3.731	3.847	3.744
80.00	3.053	2.991	3.084	2.885
85.00	2.290	2.252	2.321	2.065
90.00	1.527	1.512	1.558	1.290
95.00	0.763	0.772	0.795	0.600
100.00	0.000	0.032	0.032	0.000

FIG I-3

APPENDIX II - Mean Lines (camber profiles)

Feb 94 Sta	Camber GA-2	Camber GA-3	Camber GA-4	Camber GA-6
0.00	0.000	0.000	0.000	0.000
0.25	0.060	0.063	0.082	0.112
0.50	0.116	0.120	0.152	0.207
0.75	0.169	0.170	0.213	0.291
1.25	0.263	0.275	0.323	0.441
2.50	0.450	0.500	0.570	0.777
5.00	0.715	0.852	0.970	1.323
7.50	0.892	1.118	1.307	1.782
10.00	1.023	1.335	1.597	2.178
15.00	1.244	1.683	2.086	2.844
20.00	1.420	1.958	2.473	3.372
25.00	1.557	2.173	2.776	3.786
30.00	1.663	2.340	3.010	4.104
35.00	1.737	2.457	3.175	4.329
40.00	1.780	2.524	3.269	4.458
45.00	1.792	2.543	3.296	4.494
50.00	1.764	2.499	3.234	4.410
55.00	1.672	2.368	3.065	4.179
60.00	1.537	2.178	2.818	3.843
65.00	1.374	1.947	2.519	3.435
70.00	1.189	1.685	2.180	2.973
75.00	0.991	1.404	1.817	2.478
80.00	0.793	1.123	1.454	1.982
85.00	0.595	0.843	1.090	1.487
90.00	0.396	0.562	0.727	0.991
95.00	0.198	0.281	0.363	0.496
100.00	0.000	0.000	0.000	0.000

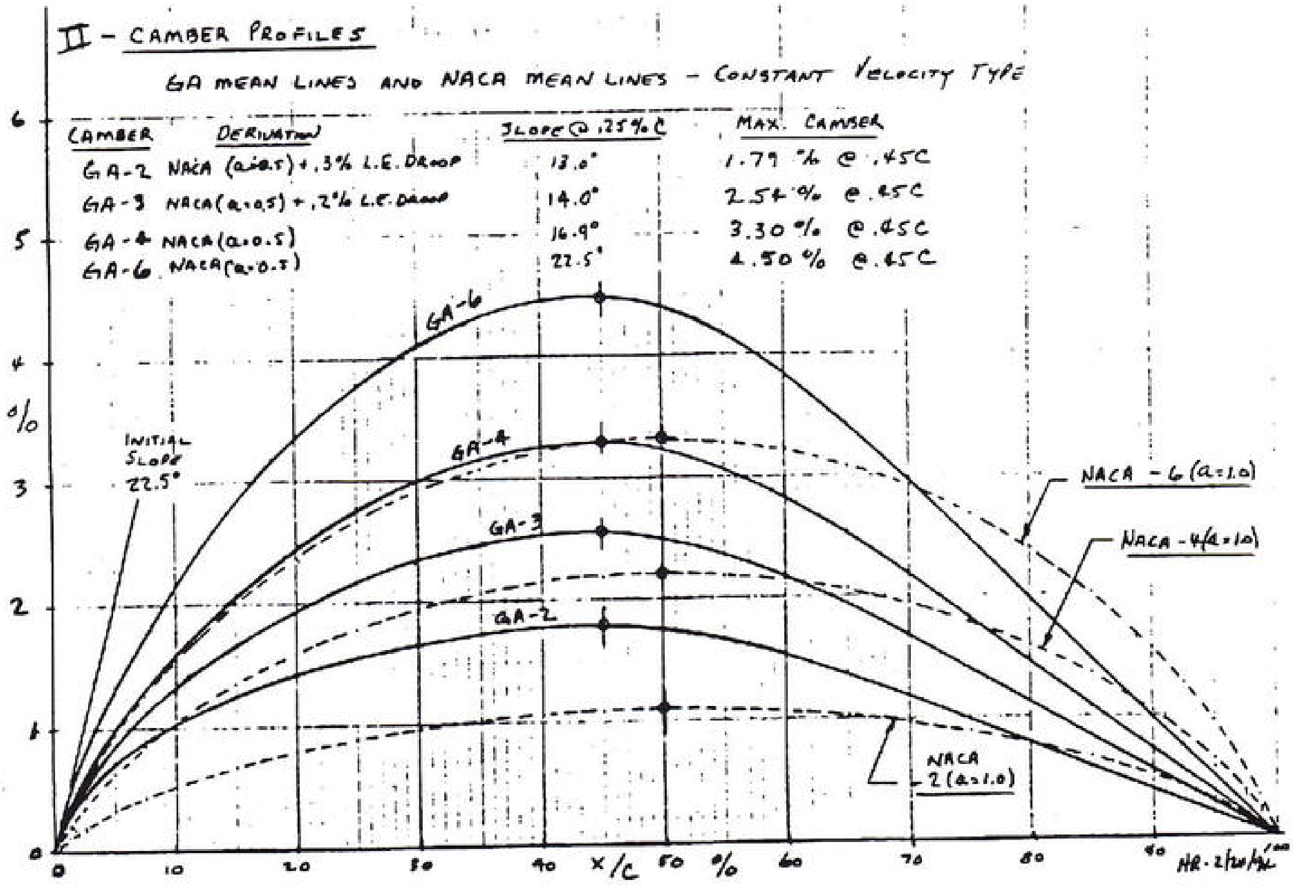
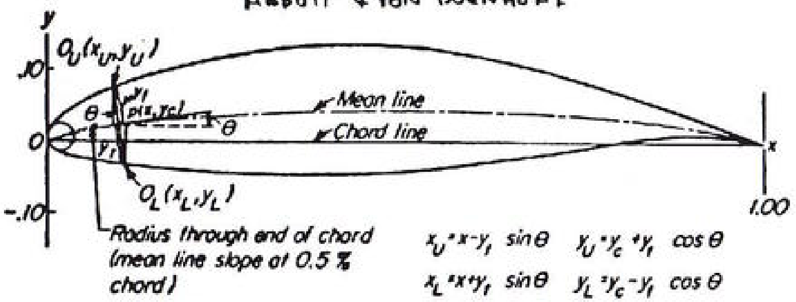


FIG II

III - Combining Mean Lines and Thickness Distributions.

(to form cambered airfoils)

112 "THEORY OF WING SECTIONS"
ABBOTT + VON DOENHOFF



$x_U = x - y_t \sin\theta$ $y_U = y_c + y_t \cos\theta$

$x_L = x + y_t \sin\theta$ $y_L = y_c - y_t \cos\theta$

Sample calculations for derivation of the NACA 65,3-818 airfoil (a = 1.0)

x	y_t*	y_c†	tan θ	sin θ	cos θ	y_t sin θ	y_t cos θ	x_U	y_U	x_L	y_L
0	0	0	....	....	....	0	0	0	0	0	0
0.005	0.01324	0.00200	0.33696‡	0.31932	0.94765	0.00423	0.01255	0.00077	0.01455	0.00923	−0.01055
0.05	0.03831	0.01264	0.18744	0.18422	0.98288	0.00706	0.03765	0.04294	0.05029	0.05706	−0.02501
0.25	0.08093	0.03580	0.06996	0.06979	0.99756	0.00565	0.08073	0.24435	0.11653	0.25565	−0.04493
0.50	0.08593	0.04412	0	0	1.00000	0	0.08593	0.50000	0.13005	0.50000	−0.04181
0.75	0.04456	0.03580	−0.06996	−0.06979	0.99756	−0.00311	0.04445	0.75311	0.08025	0.74689	−0.00865
1.00	0	0	....	....	....	0	0	1.00000	0	1.00000	0

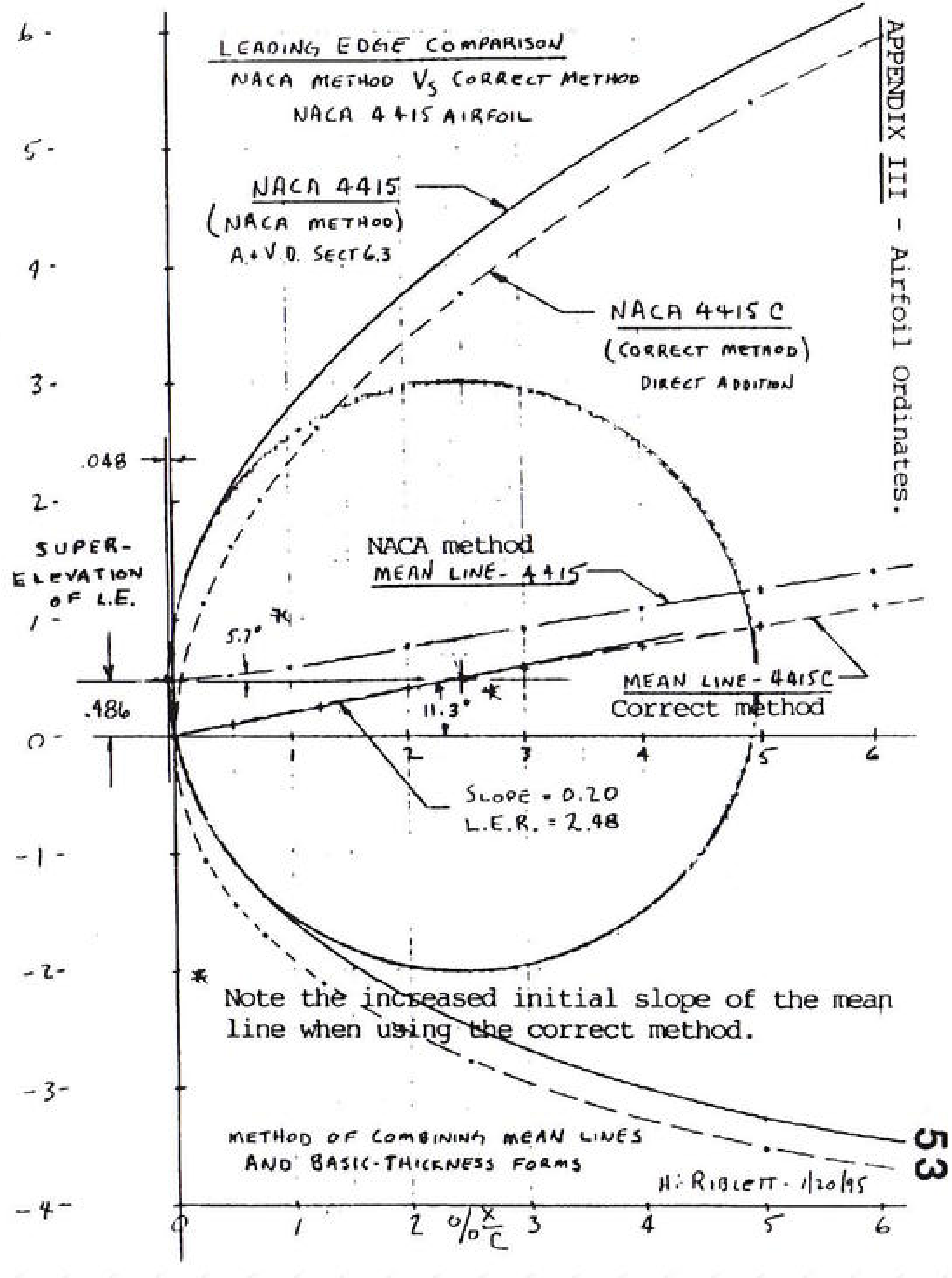
* Thickness distribution obtained from ordinates of the NACA 65,3-018 airfoil.
† Ordinates of the mean line, 0.8 of the ordinate for c_{l_i} = 1.0.
‡ Slope of radius through end of chord.

FIG. 55. Method of combining mean lines and basic-thickness forms.

6.3. Method of Combining Mean Lines and Thickness Distributions. The cambered wing sections of all NACA families of wing sections considered here are obtained by combining a mean line and a thickness distribution. The process for combining a mean line and a thickness distribution to obtain the desired cambered wing section is illustrated in Fig. 55.

This NACA method is incorrect. The correct method is simply to add the thickness distribution ordinate to the camber ordinate directly.

FIG III-1

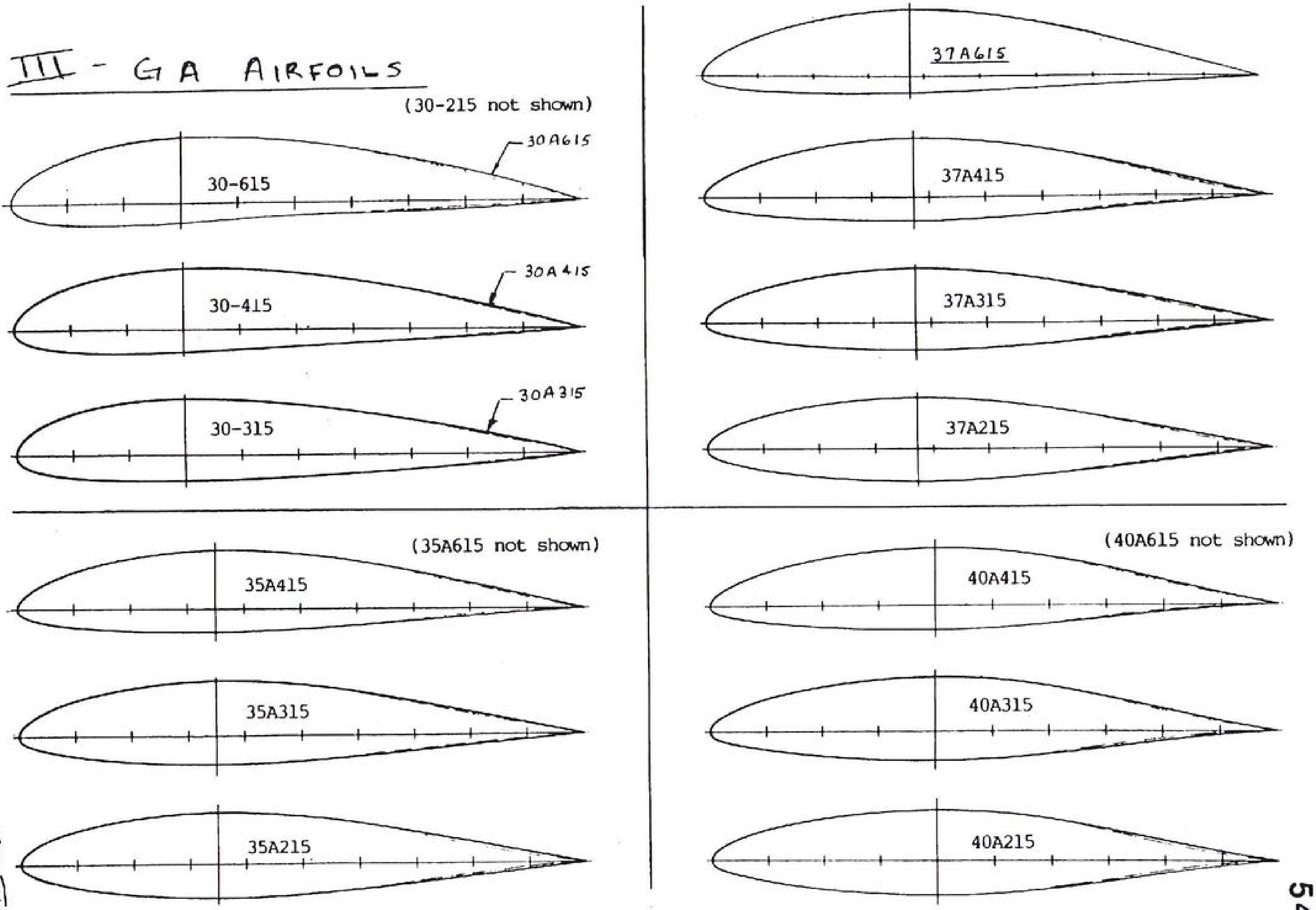


* Note the increased initial slope of the mean line when using the correct method.

METHOD OF COMBINING MEAN LINES AND BASIC-THICKNESS FORMS

H. RIBLETT · 1/20/95

III - GA AIRFOILS


(30-215 not shown)
30A615
30-615
30A415
30-415
30A315
30-315
37A615
37A415
37A315
37A215
(35A615 not shown)
35A415
35A315
35A215
(40A615 not shown)
40A415
40A315
40A215

15 % THICK GA AIRFOILS

Note- Do not scale these airfoils to change thickness, for this will also change the amount of camber in the airfoil.

FIG III-2

Feb 94	30-015	Camber	30-212		30-215		30-218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.060	0.932	-0.812	1.150	-1.030	1.368	-1.248
0.50	1.527	0.116	1.338	-1.106	1.643	-1.411	1.948	-1.716
0.75	1.856	0.169	1.654	-1.316	2.025	-1.687	2.396	-2.058
1.25	2.367	0.263	2.157	-1.631	2.630	-2.104	3.103	-2.577
2.50	3.268	0.450	3.064	-2.164	3.718	-2.818	4.372	-3.472
5.00	4.443	0.715	4.269	-2.839	5.158	-3.728	6.047	-4.617
7.50	5.250	0.892	5.092	-3.308	6.142	-4.358	7.192	-5.408
10.00	5.853	1.023	5.705	-3.659	6.876	-4.830	8.047	-6.001
15.00	6.681	1.244	6.589	-4.101	7.925	-5.437	9.261	-6.773
20.00	7.172	1.420	7.158	-4.318	8.592	-5.752	10.026	-7.186
25.00	7.427	1.557	7.499	-4.385	8.984	-5.870	10.469	-7.355
30.00	7.502	1.663	7.665	-4.339	9.165	-5.839	10.665	-7.339
35.00	7.440	1.737	7.689	-4.215	9.177	-5.703	10.665	-7.191
40.00	7.265	1.780	7.592	-4.032	9.045	-5.485	10.498	-6.938
45.00	7.000	1.792	7.392	-3.808	8.792	-5.208	10.192	-6.608
50.00	6.655	1.764	7.088	-3.560	8.419	-4.891	9.750	-6.222
55.00	6.240	1.672	6.664	-3.320	7.912	-4.568	9.160	-5.816
60.00	5.755	1.537	6.141	-3.067	7.292	-4.218	8.443	-5.369
65.00	5.190	1.374	5.526	-2.778	6.564	-3.816	7.602	-4.854
70.00	4.540	1.189	4.821	-2.443	5.729	-3.351	6.637	-4.259
75.00	3.817	0.991	4.045	-2.063	4.808	-2.826	5.571	-3.589
80.00	3.053	0.793	3.235	-1.649	3.846	-2.260	4.457	-2.871
85.00	2.290	0.595	2.427	-1.237	2.885	-1.695	3.343	-2.153
90.00	1.527	0.396	1.618	-0.826	1.923	-1.131	2.228	-1.436
95.00	0.763	0.198	0.808	-0.412	0.961	-0.565	1.114	-0.718
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	0015	Camber	30A212		30A215		30A218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.060	0.932	-0.812	1.150	-1.030	1.368	-1.248
0.50	1.527	0.116	1.338	-1.106	1.643	-1.411	1.948	-1.716
0.75	1.856	0.169	1.654	-1.316	2.025	-1.687	2.396	-2.058
1.25	2.367	0.263	2.157	-1.631	2.630	-2.104	3.103	-2.577
2.50	3.268	0.450	3.064	-2.164	3.718	-2.818	4.372	-3.472
5.00	4.443	0.715	4.269	-2.839	5.158	-3.728	6.047	-4.617
7.50	5.250	0.892	5.092	-3.308	6.142	-4.358	7.192	-5.408
10.00	5.853	1.023	5.705	-3.659	6.876	-4.830	8.047	-6.001
15.00	6.681	1.244	6.589	-4.101	7.925	-5.437	9.261	-6.773
20.00	7.172	1.420	7.158	-4.318	8.592	-5.752	10.026	-7.186
25.00	7.427	1.557	7.499	-4.385	8.984	-5.870	10.469	-7.355
30.00	7.502	1.663	7.665	-4.339	9.165	-5.839	10.665	-7.339
35.00	7.436	1.737	7.686	-4.212	9.173	-5.699	10.660	-7.186
40.00	7.254	1.780	7.583	-4.023	9.034	-5.474	10.485	-6.925
45.00	6.976	1.792	7.373	-3.789	8.768	-5.184	10.163	-6.579
50.00	6.618	1.764	7.058	-3.530	8.382	-4.854	9.706	-6.178
55.00	6.191	1.672	6.625	-3.281	7.863	-4.519	9.101	-5.757
60.00	5.704	1.537	6.100	-3.026	7.241	-4.167	8.382	-5.308
65.00	5.166	1.374	5.507	-2.759	6.540	-3.792	7.573	-4.825
70.00	4.580	1.189	4.853	-2.475	5.769	-3.391	6.685	-4.307
75.00	3.950	0.991	4.151	-2.169	4.941	-2.959	5.731	-3.749
80.00	3.279	0.793	3.416	-1.830	4.072	-2.486	4.728	-3.142
85.00	2.566	0.595	2.648	-1.458	3.161	-1.971	3.674	-2.484
90.00	1.810	0.396	1.844	-1.052	2.206	-1.414	2.568	-1.776
95.00	1.008	0.198	1.004	-0.608	1.206	-0.810	1.408	-1.012
100.00	0.158	0.000	0.126	-0.126	0.158	-0.158	0.190	-0.190

FIG III-3

Feb 96	30-015	Camber	30-312		30-315		30-318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.063	0.935	-0.809	1.153	-1.027	1.371	-1.245
0.50	1.527	0.120	1.342	-1.102	1.647	-1.407	1.952	-1.712
0.75	1.856	0.170	1.655	-1.315	2.026	-1.686	2.397	-2.057
1.25	2.367	0.275	2.169	-1.619	2.642	-2.092	3.115	-2.565
2.50	3.268	0.500	3.114	-2.114	3.768	-2.768	4.422	-3.422
5.00	4.443	0.852	4.406	-2.702	5.295	-3.591	6.184	-4.480
7.50	5.250	1.118	5.318	-3.082	6.368	-4.132	7.418	-5.182
10.00	5.853	1.335	6.017	-3.347	7.188	-4.518	8.359	-5.689
15.00	6.681	1.683	7.028	-3.662	8.364	-4.998	9.700	-6.334
20.00	7.172	1.958	7.696	-3.780	9.130	-5.214	10.564	-6.648
25.00	7.427	2.173	8.115	-3.769	9.600	-5.254	11.085	-6.739
30.00	7.502	2.340	8.342	-3.662	9.842	-5.162	11.342	-6.662
35.00	7.440	2.457	8.409	-3.495	9.897	-4.983	11.385	-6.471
40.00	7.265	2.524	8.336	-3.288	9.789	-4.741	11.242	-6.194
45.00	7.000	2.543	8.143	-3.057	9.543	-4.457	10.943	-5.857
50.00	6.655	2.499	7.823	-2.825	9.154	-4.156	10.485	-5.487
55.00	6.240	2.368	7.360	-2.624	8.608	-3.872	9.856	-5.120
60.00	5.755	2.178	6.782	-2.426	7.933	-3.577	9.084	-4.728
65.00	5.190	1.947	6.099	-2.205	7.137	-3.243	8.175	-4.281
70.00	4.540	1.685	5.317	-1.947	6.225	-2.855	7.133	-3.763
75.00	3.817	1.404	4.458	-1.650	5.221	-2.413	5.984	-3.176
80.00	3.053	1.123	3.565	-1.319	4.176	-1.930	4.787	-2.541
85.00	2.290	0.843	2.675	-0.989	3.133	-1.447	3.591	-1.905
90.00	1.527	0.562	1.784	-0.660	2.089	-0.965	2.394	-1.270
95.00	0.763	0.281	0.891	-0.329	1.044	-0.482	1.197	-0.635
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 96	0015	Camber	30A312		30A315		30A318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.063	0.935	-0.809	1.153	-1.027	1.371	-1.245
0.50	1.527	0.120	1.342	-1.102	1.647	-1.407	1.952	-1.712
0.75	1.856	0.170	1.655	-1.315	2.026	-1.686	2.397	-2.057
1.25	2.367	0.275	2.169	-1.619	2.642	-2.092	3.115	-2.565
2.50	3.268	0.500	3.114	-2.114	3.768	-2.768	4.422	-3.422
5.00	4.443	0.852	4.406	-2.702	5.295	-3.591	6.184	-4.480
7.50	5.250	1.118	5.318	-3.082	6.368	-4.132	7.418	-5.182
10.00	5.853	1.335	6.017	-3.347	7.188	-4.518	8.359	-5.689
15.00	6.681	1.683	7.028	-3.662	8.364	-4.998	9.700	-6.334
20.00	7.172	1.958	7.696	-3.780	9.130	-5.214	10.564	-6.648
25.00	7.427	2.173	8.115	-3.769	9.600	-5.254	11.085	-6.739
30.00	7.502	2.340	8.342	-3.662	9.842	-5.162	11.342	-6.662
35.00	7.436	2.457	8.406	-3.492	9.893	-4.979	11.380	-6.466
40.00	7.254	2.524	8.327	-3.279	9.778	-4.730	11.229	-6.181
45.00	6.976	2.543	8.124	-3.038	9.519	-4.433	10.914	-5.828
50.00	6.618	2.499	7.793	-2.795	9.117	-4.119	10.441	-5.443
55.00	6.191	2.368	7.321	-2.585	8.559	-3.823	9.797	-5.061
60.00	5.704	2.178	6.741	-2.385	7.882	-3.526	9.023	-4.667
65.00	5.166	1.947	6.080	-2.186	7.113	-3.219	8.146	-4.252
70.00	4.580	1.685	5.349	-1.979	6.265	-2.895	7.181	-3.811
75.00	3.950	1.404	4.564	-1.756	5.354	-2.546	6.144	-3.336
80.00	3.279	1.123	3.746	-1.500	4.402	-2.156	5.058	-2.812
85.00	2.566	0.843	2.896	-1.210	3.409	-1.723	3.922	-2.236
90.00	1.810	0.562	2.010	-0.886	2.372	-1.248	2.734	-1.610
95.00	1.008	0.281	1.087	-0.525	1.289	-0.727	1.491	-0.929
100.00	0.158	0.000	0.126	-0.126	0.158	-0.158	0.190	-0.190

FIG III-4

Feb 94	30-015	Camber	30-412		30-415		30-418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.082	0.954	-0.790	1.172	-1.008	1.390	-1.226
0.50	1.527	0.152	1.374	-1.070	1.679	-1.375	1.984	-1.680
0.75	1.856	0.213	1.698	-1.272	2.069	-1.643	2.440	-2.014
1.25	2.367	0.323	2.217	-1.571	2.690	-2.044	3.163	-2.517
2.50	3.268	0.570	3.184	-2.044	3.838	-2.698	4.492	-3.352
5.00	4.443	0.970	4.524	-2.584	5.413	-3.473	6.302	-4.362
7.50	5.250	1.307	5.507	-2.893	6.557	-3.943	7.607	-4.993
10.00	5.853	1.597	6.279	-3.085	7.450	-4.256	8.621	-5.427
15.00	6.681	2.086	7.431	-3.259	8.767	-4.595	10.103	-5.931
20.00	7.172	2.473	8.211	-3.265	9.645	-4.699	11.079	-6.133
25.00	7.427	2.776	8.718	-3.166	10.203	-4.651	11.688	-6.136
30.00	7.502	3.010	9.012	-2.992	10.512	-4.492	12.012	-5.992
35.00	7.440	3.175	9.127	-2.777	10.615	-4.265	12.103	-5.753
40.00	7.265	3.269	9.081	-2.543	10.534	-3.996	11.987	-5.449
45.00	7.000	3.296	8.896	-2.304	10.296	-3.704	11.696	-5.104
50.00	6.655	3.234	8.558	-2.090	9.889	-3.421	11.220	-4.752
55.00	6.240	3.065	8.057	-1.927	9.305	-3.175	10.553	-4.423
60.00	5.755	2.818	7.422	-1.786	8.573	-2.937	9.724	-4.088
65.00	5.190	2.519	6.671	-1.633	7.709	-2.671	8.747	-3.709
70.00	4.540	2.180	5.812	-1.452	6.720	-2.360	7.628	-3.268
75.00	3.817	1.817	4.871	-1.237	5.634	-2.000	6.397	-2.763
80.00	3.053	1.454	3.896	-0.988	4.507	-1.599	5.118	-2.210
85.00	2.290	1.090	2.922	-0.742	3.380	-1.200	3.838	-1.658
90.00	1.527	0.727	1.949	-0.495	2.254	-0.800	2.559	-1.105
95.00	0.763	0.363	0.973	-0.247	1.126	-0.400	1.279	-0.553
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	0015	Camber	30A412		30A415		30A418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.082	0.954	-0.790	1.172	-1.008	1.390	-1.226
0.50	1.527	0.152	1.374	-1.070	1.679	-1.375	1.984	-1.680
0.75	1.856	0.213	1.698	-1.272	2.069	-1.643	2.440	-2.014
1.25	2.367	0.323	2.217	-1.571	2.690	-2.044	3.163	-2.517
2.50	3.268	0.570	3.184	-2.044	3.838	-2.698	4.492	-3.352
5.00	4.443	0.970	4.524	-2.584	5.413	-3.473	6.302	-4.362
7.50	5.250	1.307	5.507	-2.893	6.557	-3.943	7.607	-4.993
10.00	5.853	1.597	6.279	-3.085	7.450	-4.256	8.621	-5.427
15.00	6.681	2.086	7.431	-3.259	8.767	-4.595	10.103	-5.931
20.00	7.172	2.473	8.211	-3.265	9.645	-4.699	11.079	-6.133
25.00	7.427	2.776	8.718	-3.166	10.203	-4.651	11.688	-6.136
30.00	7.502	3.010	9.012	-2.992	10.512	-4.492	12.012	-5.992
35.00	7.436	3.175	9.124	-2.774	10.611	-4.261	12.098	-5.748
40.00	7.254	3.269	9.072	-2.534	10.523	-3.985	11.974	-5.436
45.00	6.976	3.296	8.877	-2.285	10.272	-3.680	11.667	-5.075
50.00	6.618	3.234	8.528	-2.060	9.852	-3.384	11.176	-4.708
55.00	6.191	3.065	8.018	-1.888	9.256	-3.126	10.494	-4.364
60.00	5.704	2.818	7.381	-1.745	8.522	-2.886	9.663	-4.027
65.00	5.166	2.519	6.652	-1.614	7.685	-2.647	8.718	-3.680
70.00	4.580	2.180	5.844	-1.484	6.760	-2.400	7.676	-3.316
75.00	3.950	1.817	4.977	-1.343	5.767	-2.133	6.557	-2.923
80.00	3.279	1.454	4.077	-1.169	4.733	-1.825	5.389	-2.481
85.00	2.566	1.090	3.143	-0.963	3.656	-1.476	4.169	-1.989
90.00	1.810	0.727	2.175	-0.721	2.537	-1.083	2.899	-1.445
95.00	1.008	0.363	1.169	-0.443	1.371	-0.645	1.573	-0.847
100.00	0.158	0.000	0.126	-0.126	0.158	-0.158	0.190	-0.190

FIG III-5

Feb 94	30-015	Camber	30-612		30-615		30-618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.112	0.984	-0.760	1.202	-0.978	1.420	-1.196
0.50	1.527	0.207	1.429	-1.015	1.734	-1.320	2.039	-1.625
0.75	1.856	0.291	1.776	-1.194	2.147	-1.565	2.518	-1.936
1.25	2.367	0.441	2.335	-1.453	2.808	-1.926	3.281	-2.399
2.50	3.268	0.777	3.391	-1.837	4.045	-2.491	4.699	-3.145
5.00	4.443	1.323	4.877	-2.231	5.766	-3.120	6.655	-4.009
7.50	5.250	1.782	5.982	-2.418	7.032	-3.468	8.082	-4.518
10.00	5.853	2.178	6.860	-2.504	8.031	-3.675	9.202	-4.846
15.00	6.681	2.844	8.189	-2.501	9.525	-3.837	10.861	-5.173
20.00	7.172	3.372	9.110	-2.366	10.544	-3.800	11.978	-5.234
25.00	7.427	3.786	9.728	-2.156	11.213	-3.641	12.698	-5.126
30.00	7.502	4.104	10.106	-1.898	11.606	-3.398	13.106	-4.898
35.00	7.440	4.329	10.281	-1.623	11.769	-3.111	13.257	-4.599
40.00	7.265	4.458	10.270	-1.354	11.723	-2.807	13.176	-4.260
45.00	7.000	4.494	10.094	-1.106	11.494	-2.506	12.894	-3.906
50.00	6.655	4.410	9.734	-0.914	11.065	-2.245	12.396	-3.576
55.00	6.240	4.179	9.171	-0.813	10.419	-2.061	11.667	-3.309
60.00	5.755	3.843	8.447	-0.761	9.598	-1.912	10.749	-3.063
65.00	5.190	3.435	7.587	-0.717	8.625	-1.755	9.663	-2.793
70.00	4.540	2.973	6.605	-0.659	7.513	-1.567	8.421	-2.475
75.00	3.817	2.478	5.532	-0.576	6.295	-1.339	7.058	-2.102
80.00	3.053	1.982	4.424	-0.460	5.035	-1.071	5.646	-1.682
85.00	2.290	1.487	3.319	-0.345	3.777	-0.803	4.235	-1.261
90.00	1.527	0.991	2.213	-0.231	2.518	-0.536	2.823	-0.841
95.00	0.763	0.496	1.106	-0.114	1.259	-0.267	1.412	-0.420
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	0015	Camber	30A612		30A615		30A618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.112	0.984	-0.760	1.202	-0.978	1.420	-1.196
0.50	1.527	0.207	1.429	-1.015	1.734	-1.320	2.039	-1.625
0.75	1.856	0.291	1.776	-1.194	2.147	-1.565	2.518	-1.936
1.25	2.367	0.441	2.335	-1.453	2.808	-1.926	3.281	-2.399
2.50	3.268	0.777	3.391	-1.837	4.045	-2.491	4.699	-3.145
5.00	4.443	1.323	4.877	-2.231	5.766	-3.120	6.655	-4.009
7.50	5.250	1.782	5.982	-2.418	7.032	-3.468	8.082	-4.518
10.00	5.853	2.178	6.860	-2.504	8.031	-3.675	9.202	-4.846
15.00	6.681	2.844	8.189	-2.501	9.525	-3.837	10.861	-5.173
20.00	7.172	3.372	9.110	-2.366	10.544	-3.800	11.978	-5.234
25.00	7.427	3.786	9.728	-2.156	11.213	-3.641	12.698	-5.126
30.00	7.502	4.104	10.106	-1.898	11.606	-3.398	13.106	-4.898
35.00	7.436	4.329	10.278	-1.620	11.765	-3.107	13.252	-4.594
40.00	7.254	4.458	10.261	-1.345	11.712	-2.796	13.163	-4.247
45.00	6.976	4.494	10.075	-1.087	11.470	-2.482	12.865	-3.877
50.00	6.618	4.410	9.704	-0.884	11.028	-2.208	12.352	-3.532
55.00	6.191	4.179	9.132	-0.774	10.370	-2.012	11.608	-3.250
60.00	5.704	3.843	8.406	-0.720	9.547	-1.861	10.688	-3.002
65.00	5.166	3.435	7.568	-0.698	8.601	-1.731	9.634	-2.764
70.00	4.580	2.973	6.637	-0.691	7.553	-1.607	8.469	-2.523
75.00	3.950	2.478	5.638	-0.682	6.428	-1.472	7.218	-2.262
80.00	3.279	1.982	4.605	-0.641	5.261	-1.297	5.917	-1.953
85.00	2.566	1.487	3.540	-0.566	4.053	-1.079	4.566	-1.592
90.00	1.810	0.991	2.439	-0.457	2.801	-0.819	3.163	-1.181
95.00	1.008	0.496	1.302	-0.310	1.504	-0.512	1.706	-0.714
100.00	0.158	0.000	0.126	-0.126	0.158	-0.158	0.190	-0.190

FIG III-6

Feb 94	63-015	Camber	35-212		35-215		35-218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.060	0.760	-0.640	0.935	-0.815	1.110	-0.990
0.50	1.204	0.116	1.079	-0.847	1.320	-1.088	1.561	-1.329
0.75	1.462	0.169	1.339	-1.001	1.631	-1.293	1.923	-1.585
1.25	1.878	0.263	1.765	-1.239	2.141	-1.615	2.517	-1.991
2.50	2.610	0.450	2.538	-1.638	3.060	-2.160	3.582	-2.682
5.00	3.648	0.715	3.633	-2.203	4.363	-2.933	5.093	-3.663
7.50	4.427	0.892	4.434	-2.650	5.319	-3.535	6.204	-4.420
10.00	5.055	1.023	5.067	-3.021	6.078	-4.032	7.089	-5.043
15.00	6.011	1.244	6.053	-3.565	7.255	-4.767	8.457	-5.969
20.00	6.693	1.420	6.774	-3.934	8.113	-5.273	9.452	-6.612
25.00	7.155	1.557	7.281	-4.167	8.712	-5.598	10.143	-7.029
30.00	7.421	1.663	7.600	-4.274	9.084	-5.758	10.568	-7.242
35.00	7.500	1.737	7.737	-4.263	9.237	-5.763	10.737	-7.263
40.00	7.386	1.780	7.689	-4.129	9.166	-5.606	10.643	-7.083
45.00	7.099	1.792	7.471	-3.887	8.891	-5.307	10.311	-6.727
50.00	6.665	1.764	7.096	-3.568	8.429	-4.901	9.762	-6.234
55.00	6.108	1.672	6.558	-3.214	7.780	-4.436	9.002	-5.658
60.00	5.453	1.537	5.899	-2.825	6.990	-3.916	8.081	-5.007
65.00	4.721	1.374	5.151	-2.403	6.095	-3.347	7.039	-4.291
70.00	3.934	1.189	4.336	-1.958	5.123	-2.745	5.910	-3.532
75.00	3.119	0.991	3.486	-1.504	4.110	-2.128	4.734	-2.752
80.00	2.310	0.793	2.641	-1.055	3.103	-1.517	3.565	-1.979
85.00	1.541	0.595	1.828	-0.638	2.136	-0.946	2.444	-1.254
90.00	0.852	0.396	1.078	-0.286	1.248	-0.456	1.418	-0.626
95.00	0.300	0.198	0.438	-0.042	0.498	-0.102	0.558	-0.162
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	63A015	Camber	35A212		35A215		35A218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.060	0.760	-0.640	0.935	-0.815	1.110	-0.990
0.50	1.203	0.116	1.078	-0.846	1.319	-1.087	1.560	-1.328
0.75	1.448	0.169	1.327	-0.989	1.617	-1.279	1.907	-1.569
1.25	1.844	0.263	1.738	-1.212	2.107	-1.581	2.476	-1.950
2.50	2.579	0.450	2.513	-1.613	3.029	-2.129	3.545	-2.645
5.00	3.618	0.715	3.609	-2.179	4.333	-2.903	5.057	-3.627
7.50	4.382	0.892	4.398	-2.614	5.274	-3.490	6.150	-4.366
10.00	4.997	1.023	5.021	-2.975	6.020	-3.974	7.019	-4.973
15.00	5.942	1.244	5.998	-3.510	7.186	-4.698	8.374	-5.886
20.00	6.619	1.420	6.715	-3.875	8.039	-5.199	9.363	-6.523
25.00	7.091	1.557	7.230	-4.116	8.648	-5.534	10.066	-6.952
30.00	7.384	1.663	7.570	-4.244	9.047	-5.721	10.524	-7.198
35.00	7.496	1.737	7.734	-4.260	9.233	-5.759	10.732	-7.258
40.00	7.435	1.780	7.728	-4.168	9.215	-5.655	10.702	-7.142
45.00	7.215	1.792	7.564	-3.980	9.007	-5.423	10.450	-6.866
50.00	6.858	1.764	7.250	-3.722	8.622	-5.094	9.994	-6.466
55.00	6.387	1.672	6.782	-3.438	8.059	-4.715	9.336	-5.992
60.00	5.820	1.537	6.193	-3.119	7.357	-4.283	8.521	-5.447
65.00	5.173	1.374	5.512	-2.764	6.547	-3.799	7.582	-4.834
70.00	4.468	1.189	4.763	-2.385	5.657	-3.279	6.551	-4.173
75.00	3.731	0.991	3.976	-1.994	4.722	-2.740	5.468	-3.486
80.00	2.991	0.793	3.186	-1.600	3.784	-2.198	4.382	-2.796
85.00	2.252	0.595	2.397	-1.207	2.847	-1.657	3.297	-2.107
90.00	1.512	0.396	1.606	-0.814	1.908	-1.116	2.210	-1.418
95.00	0.772	0.198	0.816	-0.420	0.970	-0.574	1.124	-0.728
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-7

Feb 96	63-015	Camber	35-312		35-315		35-318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.063	0.763	-0.637	0.938	-0.812	1.113	-0.987
0.50	1.204	0.120	1.083	-0.843	1.324	-1.084	1.565	-1.325
0.75	1.462	0.170	1.340	-1.000	1.632	-1.292	1.924	-1.584
1.25	1.878	0.275	1.777	-1.227	2.153	-1.603	2.529	-1.979
2.50	2.610	0.500	2.588	-1.588	3.110	-2.110	3.632	-2.632
5.00	3.648	0.852	3.770	-2.066	4.500	-2.796	5.230	-3.526
7.50	4.427	1.118	4.660	-2.424	5.545	-3.309	6.430	-4.194
10.00	5.055	1.335	5.379	-2.709	6.390	-3.720	7.401	-4.731
15.00	6.011	1.683	6.492	-3.126	7.694	-4.328	8.896	-5.530
20.00	6.693	1.958	7.312	-3.396	8.651	-4.735	9.990	-6.074
25.00	7.155	2.173	7.897	-3.551	9.328	-4.982	10.759	-6.413
30.00	7.421	2.340	8.277	-3.597	9.761	-5.081	11.245	-6.565
35.00	7.500	2.457	8.457	-3.543	9.957	-5.043	11.457	-6.543
40.00	7.386	2.524	8.433	-3.385	9.910	-4.862	11.387	-6.339
45.00	7.099	2.543	8.222	-3.136	9.642	-4.556	11.062	-5.976
50.00	6.665	2.499	7.831	-2.833	9.164	-4.166	10.497	-5.499
55.00	6.108	2.368	7.254	-2.518	8.476	-3.740	9.698	-4.962
60.00	5.453	2.178	6.540	-2.184	7.631	-3.275	8.722	-4.366
65.00	4.721	1.947	5.724	-1.830	6.668	-2.774	7.612	-3.718
70.00	3.934	1.685	4.832	-1.462	5.619	-2.249	6.406	-3.036
75.00	3.119	1.404	3.899	-1.091	4.523	-1.715	5.147	-2.339
80.00	2.310	1.123	2.971	-0.725	3.433	-1.187	3.895	-1.649
85.00	1.541	0.843	2.076	-0.390	2.384	-0.698	2.692	-1.006
90.00	0.852	0.562	1.244	-0.120	1.414	-0.290	1.584	-0.460
95.00	0.300	0.281	0.521	0.041	0.581	-0.019	0.641	-0.079
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 96	63A015	Camber	35A312		35A315		35A318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.063	0.763	-0.637	0.938	-0.812	1.113	-0.987
0.50	1.203	0.120	1.082	-0.842	1.323	-1.083	1.564	-1.324
0.75	1.448	0.170	1.328	-0.988	1.618	-1.278	1.908	-1.568
1.25	1.844	0.275	1.750	-1.200	2.119	-1.569	2.488	-1.938
2.50	2.579	0.500	2.563	-1.563	3.079	-2.079	3.595	-2.595
5.00	3.618	0.852	3.746	-2.042	4.470	-2.766	5.194	-3.490
7.50	4.382	1.118	4.624	-2.388	5.500	-3.264	6.376	-4.140
10.00	4.997	1.335	5.333	-2.663	6.332	-3.662	7.331	-4.661
15.00	5.942	1.683	6.437	-3.071	7.625	-4.259	8.813	-5.447
20.00	6.619	1.958	7.253	-3.337	8.577	-4.661	9.901	-5.985
25.00	7.091	2.173	7.846	-3.500	9.264	-4.918	10.682	-6.336
30.00	7.384	2.340	8.247	-3.567	9.724	-5.044	11.201	-6.521
35.00	7.496	2.457	8.454	-3.540	9.953	-5.039	11.452	-6.538
40.00	7.435	2.524	8.472	-3.424	9.959	-4.911	11.446	-6.398
45.00	7.215	2.543	8.315	-3.229	9.758	-4.672	11.201	-6.115
50.00	6.858	2.499	7.985	-2.987	9.357	-4.359	10.729	-5.731
55.00	6.387	2.368	7.478	-2.742	8.755	-4.019	10.032	-5.296
60.00	5.820	2.178	6.834	-2.478	7.998	-3.642	9.162	-4.806
65.00	5.173	1.947	6.085	-2.191	7.120	-3.226	8.155	-4.261
70.00	4.468	1.685	5.259	-1.889	6.153	-2.783	7.047	-3.677
75.00	3.731	1.404	4.389	-1.581	5.135	-2.327	5.881	-3.073
80.00	2.991	1.123	3.516	-1.270	4.114	-1.868	4.712	-2.466
85.00	2.252	0.843	2.645	-0.959	3.095	-1.409	3.545	-1.859
90.00	1.512	0.562	1.772	-0.648	2.074	-0.950	2.376	-1.252
95.00	0.772	0.281	0.899	-0.337	1.053	-0.491	1.207	-0.645
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-8

Feb 94	63-015	Camber	35-412		35-415		35-418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.082	0.782	-0.618	0.957	-0.793	1.132	-0.968
0.50	1.204	0.152	1.115	-0.811	1.356	-1.052	1.597	-1.293
0.75	1.462	0.213	1.383	-0.956	1.675	-1.249	1.968	-1.541
1.25	1.878	0.323	1.826	-1.179	2.201	-1.555	2.577	-1.930
2.50	2.610	0.570	2.658	-1.518	3.180	-2.040	3.702	-2.562
5.00	3.648	0.970	3.889	-1.948	4.618	-2.678	5.348	-3.407
7.50	4.427	1.307	4.848	-2.235	5.734	-3.120	6.619	-4.006
10.00	5.055	1.597	5.641	-2.447	6.652	-3.458	7.663	-4.469
15.00	6.011	2.086	6.894	-2.723	8.097	-3.925	9.299	-5.128
20.00	6.693	2.473	7.827	-2.882	9.166	-4.220	10.504	-5.559
25.00	7.155	2.776	8.500	-2.948	9.931	-4.379	11.362	-5.810
30.00	7.421	3.010	8.946	-2.927	10.431	-4.411	11.915	-5.896
35.00	7.500	3.175	9.175	-2.825	10.675	-4.325	12.175	-5.825
40.00	7.386	3.269	9.178	-2.640	10.655	-4.117	12.132	-5.594
45.00	7.099	3.296	8.975	-2.384	10.395	-3.803	11.814	-5.223
50.00	6.665	3.234	8.566	-2.098	9.899	-3.431	11.232	-4.764
55.00	6.108	3.065	7.951	-1.822	9.173	-3.043	10.394	-4.265
60.00	5.453	2.818	7.181	-1.544	8.271	-2.635	9.362	-3.725
65.00	4.721	2.519	6.296	-1.258	7.240	-2.202	8.184	-3.146
70.00	3.934	2.180	5.327	-0.967	6.114	-1.754	6.901	-2.541
75.00	3.119	1.817	4.312	-0.678	4.936	-1.302	5.560	-1.926
80.00	2.310	1.454	3.302	-0.394	3.764	-0.856	4.226	-1.318
85.00	1.541	1.090	2.323	-0.142	2.631	-0.451	2.940	-0.759
90.00	0.852	0.727	1.408	0.045	1.579	-0.125	1.749	-0.296
95.00	0.300	0.363	0.603	0.123	0.663	0.063	0.723	0.003
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	63A015	Camber	35A412		35A415		35A418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.082	0.782	-0.618	0.957	-0.793	1.132	-0.968
0.50	1.203	0.152	1.114	-0.811	1.355	-1.051	1.595	-1.292
0.75	1.448	0.213	1.372	-0.945	1.661	-1.235	1.951	-1.524
1.25	1.844	0.323	1.799	-1.152	2.167	-1.521	2.536	-1.889
2.50	2.579	0.570	2.633	-1.493	3.149	-2.009	3.665	-2.525
5.00	3.618	0.970	3.865	-1.924	4.588	-2.648	5.312	-3.371
7.50	4.382	1.307	4.812	-2.199	5.689	-3.075	6.565	-3.952
10.00	4.997	1.597	5.595	-2.400	6.594	-3.400	7.594	-4.399
15.00	5.942	2.086	6.839	-2.668	8.028	-3.856	9.216	-5.045
20.00	6.619	2.473	7.768	-2.822	9.092	-4.146	10.416	-5.470
25.00	7.091	2.776	8.449	-2.896	9.867	-4.315	11.286	-5.733
30.00	7.384	3.010	8.917	-2.898	10.394	-4.374	11.870	-5.851
35.00	7.496	3.175	9.171	-2.822	10.671	-4.321	12.170	-5.821
40.00	7.435	3.269	9.217	-2.679	10.704	-4.166	12.191	-5.653
45.00	7.215	3.296	9.068	-2.476	10.511	-3.919	11.954	-5.362
50.00	6.858	3.234	8.720	-2.252	10.092	-3.624	11.464	-4.996
55.00	6.387	3.065	8.174	-2.045	9.452	-3.322	10.729	-4.600
60.00	5.820	2.818	7.474	-1.838	8.638	-3.002	9.802	-4.166
65.00	5.173	2.519	6.657	-1.619	7.692	-2.654	8.727	-3.689
70.00	4.468	2.180	5.755	-1.394	6.648	-2.288	7.542	-3.181
75.00	3.731	1.817	4.802	-1.168	5.548	-1.914	6.294	-2.660
80.00	2.991	1.454	3.847	-0.939	4.445	-1.537	5.043	-2.135
85.00	2.252	1.090	2.892	-0.711	3.342	-1.162	3.793	-1.612
90.00	1.512	0.727	1.936	-0.483	2.239	-0.785	2.541	-1.088
95.00	0.772	0.363	0.981	-0.254	1.135	-0.409	1.290	-0.563
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-9

Jul 94	63-015	Camber	35-612		35-615		35-618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.112	0.812	-0.588	0.987	-0.763	1.162	-0.938
0.50	1.204	0.207	1.170	-0.756	1.411	-0.997	1.652	-1.238
0.75	1.462	0.291	1.461	-0.879	1.753	-1.171	2.045	-1.463
1.25	1.878	0.441	1.943	-1.061	2.319	-1.437	2.695	-1.813
2.50	2.610	0.777	2.865	-1.311	3.387	-1.833	3.909	-2.355
5.00	3.648	1.323	4.241	-1.595	4.971	-2.325	5.701	-3.055
7.50	4.427	1.782	5.324	-1.760	6.209	-2.645	7.094	-3.530
10.00	5.055	2.178	6.222	-1.866	7.233	-2.877	8.244	-3.888
15.00	6.011	2.844	7.653	-1.965	8.855	-3.167	10.057	-4.369
20.00	6.693	3.372	8.726	-1.982	10.065	-3.321	11.404	-4.660
25.00	7.155	3.786	9.510	-1.938	10.941	-3.369	12.372	-4.800
30.00	7.421	4.104	10.041	-1.833	11.525	-3.317	13.009	-4.801
35.00	7.500	4.329	10.329	-1.671	11.829	-3.171	13.329	-4.671
40.00	7.386	4.458	10.367	-1.451	11.844	-2.928	13.321	-4.405
45.00	7.099	4.494	10.173	-1.185	11.593	-2.605	13.013	-4.025
50.00	6.665	4.410	9.742	-0.922	11.075	-2.255	12.408	-3.588
55.00	6.108	4.179	9.065	-0.707	10.287	-1.929	11.509	-3.151
60.00	5.453	3.843	8.205	-0.519	9.296	-1.610	10.387	-2.701
65.00	4.721	3.435	7.212	-0.342	8.156	-1.286	9.100	-2.230
70.00	3.934	2.973	6.120	-0.174	6.907	-0.961	7.694	-1.748
75.00	3.119	2.478	4.973	-0.017	5.597	-0.641	6.221	-1.265
80.00	2.310	1.982	3.830	0.134	4.292	-0.328	4.754	-0.790
85.00	1.541	1.487	2.720	0.254	3.028	-0.054	3.336	-0.362
90.00	0.852	0.991	1.673	0.309	1.843	0.139	2.013	-0.031
95.00	0.300	0.496	0.736	0.256	0.796	0.196	0.856	0.136
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Jul 94	63A015	Camber	35A612		35A615		35A618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.875	0.112	0.812	-0.588	0.987	-0.763	1.162	-0.938
0.50	1.203	0.207	1.169	-0.755	1.410	-0.996	1.651	-1.237
0.75	1.448	0.291	1.449	-0.867	1.739	-1.157	2.029	-1.447
1.25	1.844	0.441	1.916	-1.034	2.285	-1.403	2.654	-1.772
2.50	2.579	0.777	2.840	-1.286	3.356	-1.802	3.872	-2.318
5.00	3.618	1.323	4.217	-1.571	4.941	-2.295	5.665	-3.019
7.50	4.382	1.782	5.288	-1.724	6.164	-2.600	7.040	-3.476
10.00	4.997	2.178	6.176	-1.820	7.175	-2.819	8.174	-3.818
15.00	5.942	2.844	7.598	-1.910	8.786	-3.098	9.974	-4.286
20.00	6.619	3.372	8.667	-1.923	9.991	-3.247	11.315	-4.571
25.00	7.091	3.786	9.459	-1.887	10.877	-3.305	12.295	-4.723
30.00	7.384	4.104	10.011	-1.803	11.488	-3.280	12.965	-4.757
35.00	7.496	4.329	10.326	-1.668	11.825	-3.167	13.324	-4.666
40.00	7.435	4.458	10.406	-1.490	11.893	-2.977	13.380	-4.464
45.00	7.215	4.494	10.266	-1.278	11.709	-2.721	13.152	-4.164
50.00	6.858	4.410	9.896	-1.076	11.268	-2.448	12.640	-3.820
55.00	6.387	4.179	9.289	-0.931	10.566	-2.208	11.843	-3.485
60.00	5.820	3.843	8.499	-0.813	9.663	-1.977	10.827	-3.141
65.00	5.173	3.435	7.573	-0.703	8.608	-1.738	9.643	-2.773
70.00	4.468	2.973	6.547	-0.601	7.441	-1.495	8.335	-2.389
75.00	3.731	2.478	5.463	-0.507	6.209	-1.253	6.955	-1.999
80.00	2.991	1.982	4.375	-0.411	4.973	-1.009	5.571	-1.607
85.00	2.252	1.487	3.289	-0.315	3.739	-0.765	4.189	-1.215
90.00	1.512	0.991	2.201	-0.219	2.503	-0.521	2.805	-0.823
95.00	0.772	0.496	1.114	-0.122	1.268	-0.276	1.422	-0.430
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-10

Feb 94	64-015	Camber	37-212		37-215		37-218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.865	0.060	0.752	-0.632	0.925	-0.805	1.098	-0.978
0.50	1.208	0.116	1.082	-0.850	1.324	-1.092	1.566	-1.334
0.75	1.456	0.169	1.334	-0.996	1.625	-1.287	1.916	-1.578
1.25	1.842	0.263	1.737	-1.211	2.105	-1.579	2.473	-1.947
2.50	2.528	0.450	2.472	-1.572	2.978	-2.078	3.484	-2.584
5.00	3.504	0.715	3.518	-2.088	4.219	-2.789	4.920	-3.490
7.50	4.240	0.892	4.284	-2.500	5.132	-3.348	5.980	-4.196
10.00	4.842	1.023	4.897	-2.851	5.865	-3.819	6.833	-4.787
15.00	5.785	1.244	5.872	-3.384	7.029	-4.541	8.186	-5.698
20.00	6.480	1.420	6.604	-3.764	7.900	-5.060	9.196	-6.356
25.00	6.985	1.557	7.145	-4.031	8.542	-5.428	9.939	-6.825
30.00	7.319	1.663	7.518	-4.192	8.982	-5.656	10.446	-7.120
35.00	7.482	1.737	7.723	-4.249	9.219	-5.745	10.715	-7.241
40.00	7.473	1.780	7.758	-4.198	9.253	-5.693	10.748	-7.188
45.00	7.224	1.792	7.571	-3.987	9.016	-5.432	10.461	-6.877
50.00	6.810	1.764	7.212	-3.684	8.574	-5.046	9.936	-6.408
55.00	6.266	1.672	6.685	-3.341	7.938	-4.594	9.191	-5.847
60.00	5.620	1.537	6.033	-2.959	7.157	-4.083	8.281	-5.207
65.00	4.895	1.374	5.290	-2.542	6.269	-3.521	7.248	-4.500
70.00	4.113	1.189	4.479	-2.101	5.302	-2.924	6.125	-3.747
75.00	3.296	0.991	3.628	-1.646	4.287	-2.305	4.946	-2.964
80.00	2.472	0.793	2.771	-1.185	3.265	-1.679	3.759	-2.173
85.00	1.677	0.595	1.937	-0.747	2.272	-1.082	2.607	-1.417
90.00	0.950	0.396	1.156	-0.364	1.346	-0.554	1.536	-0.744
95.00	0.346	0.198	0.475	-0.079	0.544	-0.148	0.613	-0.217
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	64A015	Camber	37A212		37A215		37A218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.855	0.060	0.744	-0.624	0.915	-0.795	1.086	-0.966
0.50	1.193	0.116	1.070	-0.838	1.309	-1.077	1.548	-1.316
0.75	1.436	0.169	1.318	-0.980	1.605	-1.267	1.892	-1.554
1.25	1.815	0.263	1.715	-1.189	2.078	-1.552	2.441	-1.915
2.50	2.508	0.450	2.456	-1.556	2.958	-2.058	3.460	-2.560
5.00	3.477	0.715	3.497	-2.067	4.192	-2.762	4.887	-3.457
7.50	4.202	0.892	4.254	-2.470	5.094	-3.310	5.934	-4.150
10.00	4.799	1.023	4.862	-2.816	5.822	-3.776	6.782	-4.736
15.00	5.732	1.244	5.830	-3.342	6.976	-4.488	8.122	-5.634
20.00	6.423	1.420	6.558	-3.718	7.843	-5.003	9.128	-6.288
25.00	6.926	1.557	7.098	-3.984	8.483	-5.369	9.868	-6.754
30.00	7.270	1.663	7.479	-4.153	8.933	-5.607	10.387	-7.061
35.00	7.463	1.737	7.707	-4.233	9.200	-5.726	10.693	-7.219
40.00	7.487	1.780	7.770	-4.210	9.267	-5.707	10.764	-7.204
45.00	7.313	1.792	7.642	-4.058	9.105	-5.521	10.568	-6.984
50.00	6.978	1.764	7.346	-3.818	8.742	-5.214	10.138	-6.610
55.00	6.517	1.672	6.886	-3.542	8.189	-4.845	9.492	-6.148
60.00	5.956	1.537	6.302	-3.228	7.493	-4.419	8.684	-5.610
65.00	5.311	1.374	5.623	-2.875	6.685	-3.937	7.747	-4.999
70.00	4.600	1.189	4.869	-2.491	5.789	-3.411	6.709	-4.331
75.00	3.847	0.991	4.069	-2.087	4.838	-2.856	5.607	-3.625
80.00	3.084	0.793	3.260	-1.674	3.877	-2.291	4.494	-2.908
85.00	2.321	0.595	2.452	-1.262	2.916	-1.726	3.380	-2.190
90.00	1.558	0.396	1.642	-0.850	1.954	-1.162	2.266	-1.474
95.00	0.795	0.198	0.834	-0.438	0.993	-0.597	1.152	-0.756
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-11

Feb 96	64-015	Camber	37-312		37-315		37-318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.865	0.063	0.755	-0.629	0.928	-0.802	1.101	-0.975
0.50	1.208	0.120	1.086	-0.846	1.328	-1.088	1.570	-1.330
0.75	1.456	0.170	1.335	-0.995	1.626	-1.286	1.917	-1.577
1.25	1.842	0.275	1.749	-1.199	2.117	-1.567	2.485	-1.935
2.50	2.528	0.500	2.522	-1.522	3.028	-2.028	3.534	-2.534
5.00	3.504	0.852	3.655	-1.951	4.356	-2.652	5.057	-3.353
7.50	4.240	1.118	4.510	-2.274	5.358	-3.122	6.206	-3.970
10.00	4.842	1.335	5.209	-2.539	6.177	-3.507	7.145	-4.475
15.00	5.785	1.683	6.311	-2.945	7.468	-4.102	8.625	-5.259
20.00	6.480	1.958	7.142	-3.226	8.438	-4.522	9.734	-5.818
25.00	6.985	2.173	7.761	-3.415	9.158	-4.812	10.555	-6.209
30.00	7.319	2.340	8.195	-3.515	9.659	-4.979	11.123	-6.443
35.00	7.482	2.457	8.443	-3.529	9.939	-5.025	11.435	-6.521
40.00	7.473	2.524	8.502	-3.454	9.997	-4.949	11.492	-6.444
45.00	7.224	2.543	8.322	-3.236	9.767	-4.681	11.212	-6.126
50.00	6.810	2.499	7.947	-2.949	9.309	-4.311	10.671	-5.673
55.00	6.266	2.368	7.381	-2.645	8.634	-3.898	9.887	-5.151
60.00	5.620	2.178	6.674	-2.318	7.798	-3.442	8.922	-4.566
65.00	4.895	1.947	5.863	-1.969	6.842	-2.948	7.821	-3.927
70.00	4.113	1.685	4.975	-1.605	5.798	-2.428	6.621	-3.251
75.00	3.296	1.404	4.041	-1.233	4.700	-1.892	5.359	-2.551
80.00	2.472	1.123	3.101	-0.855	3.595	-1.349	4.089	-1.843
85.00	1.677	0.843	2.185	-0.499	2.520	-0.834	2.855	-1.169
90.00	0.950	0.562	1.322	-0.198	1.512	-0.388	1.702	-0.578
95.00	0.346	0.281	0.558	0.004	0.627	-0.065	0.696	-0.134
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 96	64A015	Camber	37A312		37A315		37A318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.855	0.063	0.747	-0.621	0.918	-0.792	1.089	-0.963
0.50	1.193	0.120	1.074	-0.834	1.313	-1.073	1.552	-1.312
0.75	1.436	0.170	1.319	-0.979	1.606	-1.266	1.893	-1.553
1.25	1.815	0.275	1.727	-1.177	2.090	-1.540	2.453	-1.903
2.50	2.508	0.500	2.506	-1.506	3.008	-2.008	3.510	-2.510
5.00	3.477	0.852	3.634	-1.930	4.329	-2.625	5.024	-3.320
7.50	4.202	1.118	4.480	-2.244	5.320	-3.084	6.160	-3.924
10.00	4.799	1.335	5.174	-2.504	6.134	-3.464	7.094	-4.424
15.00	5.732	1.683	6.269	-2.903	7.415	-4.049	8.561	-5.195
20.00	6.423	1.958	7.096	-3.180	8.381	-4.465	9.666	-5.750
25.00	6.926	2.173	7.714	-3.368	9.099	-4.753	10.484	-6.138
30.00	7.270	2.340	8.156	-3.476	9.610	-4.930	11.064	-6.384
35.00	7.463	2.457	8.427	-3.513	9.920	-5.006	11.413	-6.499
40.00	7.487	2.524	8.514	-3.466	10.011	-4.963	11.508	-6.460
45.00	7.313	2.543	8.393	-3.307	9.856	-4.770	11.319	-6.233
50.00	6.978	2.499	8.081	-3.083	9.477	-4.479	10.873	-5.875
55.00	6.517	2.368	7.582	-2.846	8.885	-4.149	10.188	-5.452
60.00	5.956	2.178	6.943	-2.587	8.134	-3.778	9.325	-4.969
65.00	5.311	1.947	6.196	-2.302	7.258	-3.364	8.320	-4.426
70.00	4.600	1.685	5.365	-1.995	6.285	-2.915	7.205	-3.835
75.00	3.847	1.404	4.482	-1.674	5.251	-2.443	6.020	-3.212
80.00	3.084	1.123	3.590	-1.344	4.207	-1.961	4.824	-2.578
85.00	2.321	0.843	2.700	-1.014	3.164	-1.478	3.628	-1.942
90.00	1.558	0.562	1.808	-0.684	2.120	-0.996	2.432	-1.308
95.00	0.795	0.281	0.917	-0.355	1.076	-0.514	1.235	-0.673
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-12

Feb 94	64-015	Camber	37-412		37-415		37-418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.865	0.082	0.774	-0.610	0.947	-0.783	1.120	-0.956
0.50	1.208	0.152	1.118	-0.815	1.360	-1.056	1.601	-1.298
0.75	1.456	0.213	1.378	-0.951	1.669	-1.243	1.961	-1.534
1.25	1.842	0.323	1.797	-1.150	2.165	-1.519	2.534	-1.887
2.50	2.528	0.570	2.592	-1.453	3.098	-1.958	3.603	-2.464
5.00	3.504	0.970	3.773	-1.833	4.474	-2.534	5.175	-3.235
7.50	4.240	1.307	4.699	-2.085	5.547	-2.933	6.395	-3.781
10.00	4.842	1.597	5.471	-2.276	6.439	-3.245	7.408	-4.213
15.00	5.785	2.086	6.714	-2.542	7.871	-3.699	9.028	-4.856
20.00	6.480	2.473	7.657	-2.711	8.953	-4.007	10.249	-5.303
25.00	6.985	2.776	8.364	-2.812	9.761	-4.209	11.158	-5.606
30.00	7.319	3.010	8.865	-2.846	10.329	-4.309	11.792	-5.773
35.00	7.482	3.175	9.160	-2.811	10.657	-4.307	12.153	-5.804
40.00	7.473	3.269	9.248	-2.709	10.742	-4.204	12.237	-5.698
45.00	7.224	3.296	9.075	-2.484	10.520	-3.928	11.964	-5.373
50.00	6.810	3.234	8.682	-2.214	10.044	-3.576	11.406	-4.938
55.00	6.266	3.065	8.077	-1.948	9.331	-3.201	10.584	-4.455
60.00	5.620	2.818	7.314	-1.678	8.438	-2.802	9.562	-3.926
65.00	4.895	2.519	6.435	-1.397	7.414	-2.376	8.393	-3.355
70.00	4.113	2.180	5.471	-1.110	6.293	-1.933	7.116	-2.755
75.00	3.296	1.817	4.454	-0.820	5.113	-1.479	5.772	-2.138
80.00	2.472	1.454	3.431	-0.524	3.926	-1.018	4.420	-1.513
85.00	1.677	1.090	2.432	-0.251	2.767	-0.587	3.103	-0.922
90.00	0.950	0.727	1.487	-0.033	1.677	-0.223	1.867	-0.413
95.00	0.346	0.363	0.640	0.087	0.709	0.017	0.779	-0.052
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	64A015	Camber	37A412		37A415		37A418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.855	0.082	0.766	-0.602	0.937	-0.773	1.108	-0.944
0.50	1.193	0.152	1.106	-0.803	1.345	-1.041	1.583	-1.280
0.75	1.436	0.213	1.362	-0.935	1.649	-1.223	1.937	-1.510
1.25	1.815	0.323	1.775	-1.129	2.138	-1.492	2.501	-1.855
2.50	2.508	0.570	2.576	-1.437	3.078	-1.938	3.579	-2.440
5.00	3.477	0.970	3.752	-1.811	4.447	-2.507	5.143	-3.202
7.50	4.202	1.307	4.668	-2.055	5.509	-2.895	6.349	-3.736
10.00	4.799	1.597	5.436	-2.242	6.396	-3.202	7.356	-4.162
15.00	5.732	2.086	6.671	-2.500	7.818	-3.646	8.964	-4.793
20.00	6.423	2.473	7.611	-2.666	8.896	-3.950	10.180	-5.235
25.00	6.926	2.776	8.317	-2.764	9.702	-4.150	11.088	-5.535
30.00	7.270	3.010	8.826	-2.806	10.280	-4.260	11.734	-5.714
35.00	7.463	3.175	9.145	-2.796	10.638	-4.288	12.130	-5.781
40.00	7.487	3.269	9.259	-2.720	10.756	-4.218	12.254	-5.715
45.00	7.313	3.296	9.146	-2.555	10.609	-4.017	12.071	-5.480
50.00	6.978	3.234	8.816	-2.348	10.212	-3.744	11.608	-5.140
55.00	6.517	3.065	8.278	-2.149	9.582	-3.452	10.885	-4.756
60.00	5.956	2.818	7.583	-1.947	8.774	-3.138	9.965	-4.329
65.00	5.311	2.519	6.768	-1.730	7.830	-2.792	8.892	-3.854
70.00	4.600	2.180	5.860	-1.500	6.780	-2.420	7.700	-3.340
75.00	3.847	1.817	4.895	-1.260	5.664	-2.030	6.434	-2.799
80.00	3.084	1.454	3.921	-1.013	4.538	-1.630	5.155	-2.247
85.00	2.321	1.090	2.947	-0.766	3.411	-1.231	3.876	-1.695
90.00	1.558	0.727	1.973	-0.520	2.285	-0.831	2.596	-1.143
95.00	0.795	0.363	0.999	-0.273	1.158	-0.432	1.317	-0.591
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-13

Feb 94	64-015	Camber	37-612		37-615		37-618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.865	0.112	0.804	-0.580	0.977	-0.753	1.150	-0.926
0.50	1.208	0.207	1.173	-0.759	1.415	-1.001	1.657	-1.243
0.75	1.456	0.291	1.456	-0.874	1.747	-1.165	2.038	-1.456
1.25	1.842	0.441	1.915	-1.033	2.283	-1.401	2.651	-1.769
2.50	2.528	0.777	2.799	-1.245	3.305	-1.751	3.811	-2.257
5.00	3.504	1.323	4.126	-1.480	4.827	-2.181	5.528	-2.882
7.50	4.240	1.782	5.174	-1.610	6.022	-2.458	6.870	-3.306
10.00	4.842	2.178	6.052	-1.696	7.020	-2.664	7.988	-3.632
15.00	5.785	2.844	7.472	-1.784	8.629	-2.941	9.786	-4.098
20.00	6.480	3.372	8.556	-1.812	9.852	-3.108	11.148	-4.404
25.00	6.985	3.786	9.374	-1.802	10.771	-3.199	12.168	-4.596
30.00	7.319	4.104	9.959	-1.751	11.423	-3.215	12.887	-4.679
35.00	7.482	4.329	10.315	-1.657	11.811	-3.153	13.307	-4.649
40.00	7.473	4.458	10.436	-1.520	11.931	-3.015	13.426	-4.510
45.00	7.224	4.494	10.273	-1.285	11.718	-2.730	13.163	-4.175
50.00	6.810	4.410	9.858	-1.038	11.220	-2.400	12.582	-3.762
55.00	6.266	4.179	9.192	-0.834	10.445	-2.087	11.698	-3.340
60.00	5.620	3.843	8.339	-0.653	9.463	-1.777	10.587	-2.901
65.00	4.895	3.435	7.351	-0.481	8.330	-1.460	9.309	-2.439
70.00	4.113	2.973	6.263	-0.317	7.086	-1.140	7.909	-1.963
75.00	3.296	2.478	5.115	-0.159	5.774	-0.818	6.433	-1.477
80.00	2.472	1.982	3.960	0.004	4.454	-0.490	4.948	-0.984
85.00	1.677	1.487	2.829	0.145	3.164	-0.190	3.499	-0.525
90.00	0.950	0.991	1.751	0.231	1.941	0.041	2.131	-0.149
95.00	0.346	0.496	0.773	0.219	0.842	0.150	0.911	0.081
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	64A015	Camber	37A612		37A615		37A618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.855	0.112	0.796	-0.572	0.967	-0.743	1.138	-0.914
0.50	1.193	0.207	1.161	-0.747	1.400	-0.986	1.639	-1.225
0.75	1.436	0.291	1.440	-0.858	1.727	-1.145	2.014	-1.432
1.25	1.815	0.441	1.893	-1.011	2.256	-1.374	2.619	-1.737
2.50	2.508	0.777	2.783	-1.229	3.285	-1.731	3.787	-2.233
5.00	3.477	1.323	4.105	-1.459	4.800	-2.154	5.495	-2.849
7.50	4.202	1.782	5.144	-1.580	5.984	-2.420	6.824	-3.260
10.00	4.799	2.178	6.017	-1.661	6.977	-2.621	7.937	-3.581
15.00	5.732	2.844	7.430	-1.742	8.576	-2.888	9.722	-4.034
20.00	6.423	3.372	8.510	-1.766	9.795	-3.051	11.080	-4.336
25.00	6.926	3.786	9.327	-1.755	10.712	-3.140	12.097	-4.525
30.00	7.270	4.104	9.920	-1.712	11.374	-3.166	12.828	-4.620
35.00	7.463	4.329	10.299	-1.641	11.792	-3.134	13.285	-4.627
40.00	7.487	4.458	10.448	-1.532	11.945	-3.029	13.442	-4.526
45.00	7.313	4.494	10.344	-1.356	11.807	-2.819	13.270	-4.282
50.00	6.978	4.410	9.992	-1.172	11.388	-2.568	12.784	-3.964
55.00	6.517	4.179	9.393	-1.035	10.696	-2.338	11.999	-3.641
60.00	5.956	3.843	8.608	-0.922	9.799	-2.113	10.990	-3.304
65.00	5.311	3.435	7.684	-0.814	8.746	-1.876	9.808	-2.938
70.00	4.600	2.973	6.653	-0.707	7.573	-1.627	8.493	-2.547
75.00	3.847	2.478	5.556	-0.600	6.325	-1.369	7.094	-2.138
80.00	3.084	1.982	4.449	-0.485	5.066	-1.102	5.683	-1.719
85.00	2.321	1.487	3.344	-0.370	3.808	-0.834	4.272	-1.298
90.00	1.558	0.991	2.237	-0.255	2.549	-0.567	2.861	-0.879
95.00	0.795	0.496	1.132	-0.140	1.291	-0.299	1.450	-0.458
100.00	0.032	0.000	0.026	-0.026	0.032	-0.032	0.038	-0.038

FIG III-14

Feb 94	65-015	Camber	40-212		40-215		40-218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.060	0.724	-0.604	0.890	-0.770	1.056	-0.936
0.50	1.124	0.116	1.015	-0.783	1.240	-1.008	1.465	-1.233
0.75	1.356	0.169	1.254	-0.916	1.525	-1.187	1.796	-1.458
1.25	1.702	0.263	1.625	-1.099	1.965	-1.439	2.305	-1.779
2.50	2.324	0.450	2.309	-1.409	2.774	-1.874	3.239	-2.339
5.00	3.245	0.715	3.311	-1.881	3.960	-2.530	4.609	-3.179
7.50	3.959	0.892	4.059	-2.275	4.851	-3.067	5.643	-3.859
10.00	4.555	1.023	4.667	-2.621	5.578	-3.532	6.489	-4.443
15.00	5.504	1.244	5.647	-3.159	6.748	-4.260	7.849	-5.361
20.00	6.223	1.420	6.398	-3.558	7.643	-4.803	8.888	-6.048
25.00	6.764	1.557	6.968	-3.854	8.321	-5.207	9.674	-6.560
30.00	7.152	1.663	7.385	-4.059	8.815	-5.489	10.245	-6.919
35.00	7.396	1.737	7.654	-4.180	9.133	-5.659	10.612	-7.138
40.00	7.498	1.780	7.778	-4.218	9.278	-5.718	10.778	-7.218
45.00	7.427	1.792	7.734	-4.150	9.219	-5.635	10.704	-7.120
50.00	7.168	1.764	7.498	-3.970	8.932	-5.404	10.366	-6.838
55.00	6.720	1.672	7.048	-3.704	8.392	-5.048	9.736	-6.392
60.00	6.118	1.537	6.431	-3.357	7.655	-4.581	8.879	-5.805
65.00	5.403	1.374	5.696	-2.948	6.777	-4.029	7.858	-5.110
70.00	4.600	1.189	4.869	-2.491	5.789	-3.411	6.709	-4.331
75.00	3.744	0.991	3.986	-2.004	4.735	-2.753	5.484	-3.502
80.00	2.858	0.793	3.079	-1.493	3.651	-2.065	4.223	-2.637
85.00	1.977	0.595	2.177	-0.987	2.572	-1.382	2.967	-1.777
90.00	1.144	0.396	1.311	-0.519	1.540	-0.748	1.769	-0.977
95.00	0.428	0.198	0.540	-0.144	0.626	-0.230	0.712	-0.316
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	40A015	Camber	40A212		40A215		40A218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.060	0.724	-0.604	0.890	-0.770	1.056	-0.936
0.50	1.124	0.116	1.015	-0.783	1.240	-1.008	1.465	-1.233
0.75	1.356	0.169	1.254	-0.916	1.525	-1.187	1.796	-1.458
1.25	1.702	0.263	1.625	-1.099	1.965	-1.439	2.305	-1.779
2.50	2.324	0.450	2.309	-1.409	2.774	-1.874	3.239	-2.339
5.00	3.245	0.715	3.311	-1.881	3.960	-2.530	4.609	-3.179
7.50	3.959	0.892	4.059	-2.275	4.851	-3.067	5.643	-3.859
10.00	4.555	1.023	4.667	-2.621	5.578	-3.532	6.489	-4.443
15.00	5.504	1.244	5.647	-3.159	6.748	-4.260	7.849	-5.361
20.00	6.223	1.420	6.398	-3.558	7.643	-4.803	8.888	-6.048
25.00	6.764	1.557	6.968	-3.854	8.321	-5.207	9.674	-6.560
30.00	7.152	1.663	7.385	-4.059	8.815	-5.489	10.245	-6.919
35.00	7.396	1.737	7.654	-4.180	9.133	-5.659	10.612	-7.138
40.00	7.498	1.780	7.778	-4.218	9.278	-5.718	10.778	-7.218
45.00	7.427	1.792	7.734	-4.150	9.219	-5.635	10.704	-7.120
50.00	7.168	1.764	7.498	-3.970	8.932	-5.404	10.366	-6.838
55.00	6.720	1.672	7.048	-3.704	8.392	-5.048	9.736	-6.392
60.00	6.118	1.537	6.431	-3.357	7.655	-4.581	8.879	-5.805
65.00	5.403	1.374	5.696	-2.948	6.777	-4.029	7.858	-5.110
70.00	4.600	1.189	4.869	-2.491	5.789	-3.411	6.709	-4.331
75.00	3.744	0.991	3.986	-2.004	4.735	-2.753	5.484	-3.502
80.00	2.885	0.793	3.101	-1.515	3.678	-2.092	4.255	-2.669
85.00	2.065	0.595	2.247	-1.057	2.660	-1.470	3.073	-1.883
90.00	1.290	0.396	1.428	-0.636	1.686	-0.894	1.944	-1.152
95.00	0.600	0.198	0.678	-0.282	0.798	-0.402	0.918	-0.522
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

FIG III-15

Feb 96	65-015	Camber	40-312		40-315		40-318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.063	0.727	-0.601	0.893	-0.767	1.059	-0.933
0.50	1.124	0.120	1.019	-0.779	1.244	-1.004	1.469	-1.229
0.75	1.356	0.170	1.255	-0.915	1.526	-1.186	1.797	-1.457
1.25	1.702	0.275	1.637	-1.087	1.977	-1.427	2.317	-1.767
2.50	2.324	0.500	2.359	-1.359	2.824	-1.824	3.289	-2.289
5.00	3.245	0.852	3.448	-1.744	4.097	-2.393	4.746	-3.042
7.50	3.959	1.118	4.285	-2.049	5.077	-2.841	5.869	-3.633
10.00	4.555	1.335	4.979	-2.309	5.890	-3.220	6.801	-4.131
15.00	5.504	1.683	6.086	-2.720	7.187	-3.821	8.288	-4.922
20.00	6.223	1.958	6.936	-3.020	8.181	-4.265	9.426	-5.510
25.00	6.764	2.173	7.584	-3.238	8.937	-4.591	10.290	-5.944
30.00	7.152	2.340	8.062	-3.382	9.492	-4.812	10.922	-6.242
35.00	7.396	2.457	8.374	-3.460	9.853	-4.939	11.332	-6.418
40.00	7.498	2.524	8.522	-3.474	10.022	-4.974	11.522	-6.474
45.00	7.427	2.543	8.485	-3.399	9.970	-4.884	11.455	-6.369
50.00	7.168	2.499	8.233	-3.235	9.667	-4.669	11.101	-6.103
55.00	6.720	2.368	7.744	-3.008	9.088	-4.352	10.432	-5.696
60.00	6.118	2.178	7.072	-2.716	8.296	-3.940	9.520	-5.164
65.00	5.403	1.947	6.269	-2.375	7.350	-3.456	8.431	-4.537
70.00	4.600	1.685	5.365	-1.995	6.285	-2.915	7.205	-3.835
75.00	3.744	1.404	4.399	-1.591	5.148	-2.340	5.897	-3.089
80.00	2.858	1.123	3.409	-1.163	3.981	-1.735	4.553	-2.307
85.00	1.977	0.843	2.425	-0.739	2.820	-1.134	3.215	-1.529
90.00	1.144	0.562	1.477	-0.353	1.706	-0.582	1.935	-0.811
95.00	0.428	0.281	0.623	-0.061	0.709	-0.147	0.795	-0.233
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 96	40A015	Camber	40A312		40A315		40A318	
Sta	Ord	GA-3	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.063	0.727	-0.601	0.893	-0.767	1.059	-0.933
0.50	1.124	0.122	1.021	-0.777	1.246	-1.002	1.471	-1.227
0.75	1.356	0.170	1.255	-0.915	1.526	-1.186	1.797	-1.457
1.25	1.702	0.275	1.637	-1.087	1.977	-1.427	2.317	-1.767
2.50	2.324	0.500	2.359	-1.359	2.824	-1.824	3.289	-2.289
5.00	3.245	0.852	3.448	-1.744	4.097	-2.393	4.746	-3.042
7.50	3.959	1.118	4.285	-2.049	5.077	-2.841	5.869	-3.633
10.00	4.555	1.335	4.979	-2.309	5.890	-3.220	6.801	-4.131
15.00	5.504	1.683	6.086	-2.720	7.187	-3.821	8.288	-4.922
20.00	6.223	1.958	6.936	-3.020	8.181	-4.265	9.426	-5.510
25.00	6.764	2.173	7.584	-3.238	8.937	-4.591	10.290	-5.944
30.00	7.152	2.340	8.062	-3.382	9.492	-4.812	10.922	-6.242
35.00	7.396	2.457	8.374	-3.460	9.853	-4.939	11.332	-6.418
40.00	7.498	2.524	8.522	-3.474	10.022	-4.974	11.522	-6.474
45.00	7.427	2.543	8.485	-3.399	9.970	-4.884	11.455	-6.369
50.00	7.168	2.499	8.233	-3.235	9.667	-4.669	11.101	-6.103
55.00	6.720	2.368	7.744	-3.008	9.088	-4.352	10.432	-5.696
60.00	6.118	2.178	7.072	-2.716	8.296	-3.940	9.520	-5.164
65.00	5.403	1.947	6.269	-2.375	7.350	-3.456	8.431	-4.537
70.00	4.600	1.685	5.365	-1.995	6.285	-2.915	7.205	-3.835
75.00	3.744	1.404	4.399	-1.591	5.148	-2.340	5.897	-3.089
80.00	2.885	1.123	3.431	-1.185	4.008	-1.762	4.585	-2.339
85.00	2.065	0.843	2.495	-0.809	2.908	-1.222	3.321	-1.635
90.00	1.290	0.562	1.594	-0.470	1.852	-0.728	2.110	-0.986
95.00	0.600	0.281	0.761	-0.199	0.881	-0.319	1.001	-0.439
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

FIG III-16

Feb 94	65-015	Camber	40-412		40-415		40-418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.082	0.746	-0.582	0.912	-0.748	1.078	-0.914
0.50	1.124	0.152	1.051	-0.747	1.276	-0.972	1.501	-1.197
0.75	1.356	0.213	1.298	-0.871	1.569	-1.143	1.841	-1.414
1.25	1.702	0.323	1.685	-1.038	2.025	-1.379	2.366	-1.719
2.50	2.324	0.570	2.429	-1.289	2.894	-1.754	3.359	-2.219
5.00	3.245	0.970	3.566	-1.626	4.215	-2.275	4.864	-2.924
7.50	3.959	1.307	4.474	-1.860	5.266	-2.652	6.058	-3.444
10.00	4.555	1.597	5.241	-2.047	6.152	-2.958	7.063	-3.869
15.00	5.504	2.086	6.489	-2.318	7.590	-3.418	8.690	-4.519
20.00	6.223	2.473	7.451	-2.506	8.696	-3.750	9.940	-4.995
25.00	6.764	2.776	8.188	-2.635	9.540	-3.988	10.893	-5.340
30.00	7.152	3.010	8.731	-2.712	10.162	-4.142	11.592	-5.573
35.00	7.396	3.175	9.091	-2.742	10.571	-4.221	12.050	-5.701
40.00	7.498	3.269	9.268	-2.729	10.767	-4.229	12.267	-5.728
45.00	7.427	3.296	9.237	-2.646	10.723	-4.131	12.208	-5.617
50.00	7.168	3.234	8.968	-2.500	10.402	-3.934	11.836	-5.368
55.00	6.720	3.065	8.441	-2.311	9.785	-3.655	11.129	-4.999
60.00	6.118	2.818	7.713	-2.076	8.936	-3.300	10.160	-4.523
65.00	5.403	2.519	6.841	-1.803	7.922	-2.884	9.003	-3.965
70.00	4.600	2.180	5.860	-1.500	6.780	-2.420	7.700	-3.340
75.00	3.744	1.817	4.812	-1.178	5.561	-1.927	6.310	-2.676
80.00	2.858	1.454	3.740	-0.833	4.312	-1.404	4.883	-1.976
85.00	1.977	1.090	2.672	-0.491	3.067	-0.887	3.463	-1.282
90.00	1.144	0.727	1.642	-0.188	1.871	-0.417	2.100	-0.646
95.00	0.428	0.363	0.706	0.021	0.791	-0.065	0.877	-0.150
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Feb 94	40A015	Camber	40A412		40A415		40A418	
Sta	Ord	GA-4	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.082	0.746	-0.582	0.912	-0.748	1.078	-0.914
0.50	1.124	0.152	1.051	-0.747	1.276	-0.972	1.501	-1.197
0.75	1.356	0.213	1.298	-0.871	1.569	-1.143	1.841	-1.414
1.25	1.702	0.323	1.685	-1.038	2.025	-1.379	2.366	-1.719
2.50	2.324	0.570	2.429	-1.289	2.894	-1.754	3.359	-2.219
5.00	3.245	0.970	3.566	-1.626	4.215	-2.275	4.864	-2.924
7.50	3.959	1.307	4.474	-1.860	5.266	-2.652	6.058	-3.444
10.00	4.555	1.597	5.241	-2.047	6.152	-2.958	7.063	-3.869
15.00	5.504	2.086	6.489	-2.318	7.590	-3.418	8.690	-4.519
20.00	6.223	2.473	7.451	-2.506	8.696	-3.750	9.940	-4.995
25.00	6.764	2.776	8.188	-2.635	9.540	-3.988	10.893	-5.340
30.00	7.152	3.010	8.731	-2.712	10.162	-4.142	11.592	-5.573
35.00	7.396	3.175	9.091	-2.742	10.571	-4.221	12.050	-5.701
40.00	7.498	3.269	9.268	-2.729	10.767	-4.229	12.267	-5.728
45.00	7.427	3.296	9.237	-2.646	10.723	-4.131	12.208	-5.617
50.00	7.168	3.234	8.968	-2.500	10.402	-3.934	11.836	-5.368
55.00	6.720	3.065	8.441	-2.311	9.785	-3.655	11.129	-4.999
60.00	6.118	2.818	7.713	-2.076	8.936	-3.300	10.160	-4.523
65.00	5.403	2.519	6.841	-1.803	7.922	-2.884	9.003	-3.965
70.00	4.600	2.180	5.860	-1.500	6.780	-2.420	7.700	-3.340
75.00	3.744	1.817	4.812	-1.178	5.561	-1.927	6.310	-2.676
80.00	2.885	1.454	3.762	-0.854	4.339	-1.431	4.916	-2.008
85.00	2.065	1.090	2.742	-0.562	3.155	-0.975	3.568	-1.388
90.00	1.290	0.727	1.759	-0.305	2.017	-0.563	2.275	-0.821
95.00	0.600	0.363	0.843	-0.117	0.963	-0.237	1.083	-0.357
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

FIG III-17

Jul 94	65-015	Camber	40-612		40-615		40-618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.112	0.776	-0.552	0.942	-0.718	1.108	-0.884
0.50	1.124	0.207	1.106	-0.692	1.331	-0.917	1.556	-1.142
0.75	1.356	0.291	1.376	-0.794	1.647	-1.065	1.918	-1.336
1.25	1.702	0.441	1.803	-0.921	2.143	-1.261	2.483	-1.601
2.50	2.324	0.777	2.636	-1.082	3.101	-1.547	3.566	-2.012
5.00	3.245	1.323	3.919	-1.273	4.568	-1.922	5.217	-2.571
7.50	3.959	1.782	4.949	-1.385	5.741	-2.177	6.533	-2.969
10.00	4.555	2.178	5.822	-1.466	6.733	-2.377	7.644	-3.288
15.00	5.504	2.844	7.247	-1.559	8.348	-2.660	9.449	-3.761
20.00	6.223	3.372	8.350	-1.606	9.595	-2.851	10.840	-4.096
25.00	6.764	3.786	9.197	-1.625	10.550	-2.978	11.903	-4.331
30.00	7.152	4.104	9.826	-1.618	11.256	-3.048	12.686	-4.478
35.00	7.396	4.329	10.246	-1.588	11.725	-3.067	13.204	-4.546
40.00	7.498	4.458	10.456	-1.540	11.956	-3.040	13.456	-4.540
45.00	7.427	4.494	10.436	-1.448	11.921	-2.933	13.406	-4.418
50.00	7.168	4.410	10.144	-1.324	11.578	-2.758	13.012	-4.192
55.00	6.720	4.179	9.555	-1.197	10.899	-2.541	12.243	-3.885
60.00	6.118	3.843	8.737	-1.051	9.961	-2.275	11.185	-3.499
65.00	5.403	3.435	7.757	-0.887	8.838	-1.968	9.919	-3.049
70.00	4.600	2.973	6.653	-0.707	7.573	-1.627	8.493	-2.547
75.00	3.744	2.478	5.473	-0.517	6.222	-1.266	6.971	-2.015
80.00	2.858	1.982	4.268	-0.304	4.840	-0.876	5.412	-1.448
85.00	1.977	1.487	3.069	-0.095	3.464	-0.490	3.859	-0.885
90.00	1.144	0.991	1.906	0.076	2.135	-0.153	2.364	-0.382
95.00	0.428	0.496	0.838	0.154	0.924	0.068	1.010	-0.018
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Jul 94	40A015	Camber	40A612		40A615		40A618	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.830	0.112	0.776	-0.552	0.942	-0.718	1.108	-0.884
0.50	1.124	0.207	1.106	-0.692	1.331	-0.917	1.556	-1.142
0.75	1.356	0.291	1.376	-0.794	1.647	-1.065	1.918	-1.336
1.25	1.702	0.441	1.803	-0.921	2.143	-1.261	2.483	-1.601
2.50	2.324	0.777	2.636	-1.082	3.101	-1.547	3.566	-2.012
5.00	3.245	1.323	3.919	-1.273	4.568	-1.922	5.217	-2.571
7.50	3.959	1.782	4.949	-1.385	5.741	-2.177	6.533	-2.969
10.00	4.555	2.178	5.822	-1.466	6.733	-2.377	7.644	-3.288
15.00	5.504	2.844	7.247	-1.559	8.348	-2.660	9.449	-3.761
20.00	6.223	3.372	8.350	-1.606	9.595	-2.851	10.840	-4.096
25.00	6.764	3.786	9.197	-1.625	10.550	-2.978	11.903	-4.331
30.00	7.152	4.104	9.826	-1.618	11.256	-3.048	12.686	-4.478
35.00	7.396	4.329	10.246	-1.588	11.725	-3.067	13.204	-4.546
40.00	7.498	4.458	10.456	-1.540	11.956	-3.040	13.456	-4.540
45.00	7.427	4.494	10.436	-1.448	11.921	-2.933	13.406	-4.418
50.00	7.168	4.410	10.144	-1.324	11.578	-2.758	13.012	-4.192
55.00	6.720	4.179	9.555	-1.197	10.899	-2.541	12.243	-3.885
60.00	6.118	3.843	8.737	-1.051	9.961	-2.275	11.185	-3.499
65.00	5.403	3.435	7.757	-0.887	8.838	-1.968	9.919	-3.049
70.00	4.600	2.973	6.653	-0.707	7.573	-1.627	8.493	-2.547
75.00	3.744	2.478	5.473	-0.517	6.222	-1.266	6.971	-2.015
80.00	2.885	1.982	4.290	-0.326	4.867	-0.903	5.444	-1.480
85.00	2.065	1.487	3.139	-0.165	3.552	-0.578	3.965	-0.991
90.00	1.290	0.991	2.023	-0.041	2.281	-0.299	2.539	-0.557
95.00	0.600	0.496	0.976	0.016	1.096	-0.104	1.216	-0.224
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

FIG III-18

HIGH LIFT AIRFOILS (see also figure III-6)

Feb 94	30-015	Camber	30-613.5		30-615		30-616.5	
Sta	Ord	GA-6	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	1.090	0.112	1.093	-0.869	1.202	-0.978	1.311	-1.087
0.50	1.527	0.207	1.581	-1.167	1.734	-1.320	1.887	-1.473
0.75	1.856	0.291	1.961	-1.379	2.147	-1.565	2.333	-1.751
1.25	2.367	0.441	2.571	-1.689	2.808	-1.926	3.045	-2.163
2.50	3.268	0.777	3.718	-2.164	4.045	-2.491	4.372	-2.818
5.00	4.443	1.323	5.322	-2.676	5.766	-3.120	6.210	-3.564
7.50	5.250	1.782	6.507	-2.943	7.032	-3.468	7.557	-3.993
10.00	5.853	2.178	7.446	-3.090	8.031	-3.675	8.616	-4.260
15.00	6.681	2.844	8.857	-3.169	9.525	-3.837	10.193	-4.505
20.00	7.172	3.372	9.827	-3.083	10.544	-3.800	11.261	-4.517
25.00	7.427	3.786	10.470	-2.898	11.213	-3.641	11.956	-4.384
30.00	7.502	4.104	10.856	-2.648	11.606	-3.398	12.356	-4.148
35.00	7.440	4.329	11.025	-2.367	11.769	-3.111	12.513	-3.855
40.00	7.265	4.458	10.997	-2.080	11.723	-2.807	12.450	-3.533
45.00	7.000	4.494	10.794	-1.806	11.494	-2.506	12.194	-3.206
50.00	6.655	4.410	10.400	-1.580	11.065	-2.245	11.731	-2.911
55.00	6.240	4.179	9.795	-1.437	10.419	-2.061	11.043	-2.685
60.00	5.755	3.843	9.023	-1.337	9.598	-1.912	10.174	-2.488
65.00	5.190	3.435	8.106	-1.236	8.625	-1.755	9.144	-2.274
70.00	4.540	2.973	7.059	-1.113	7.513	-1.567	7.967	-2.021
75.00	3.817	2.478	5.913	-0.957	6.295	-1.339	6.677	-1.721
80.00	3.053	1.982	4.730	-0.766	5.035	-1.071	5.340	-1.376
85.00	2.290	1.487	3.548	-0.574	3.777	-0.803	4.006	-1.032
90.00	1.527	0.991	2.365	-0.383	2.518	-0.536	2.671	-0.689
95.00	0.763	0.496	1.183	-0.191	1.259	-0.267	1.335	-0.343
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000

Table 1 - Predicted Performance Summary (Computer analysis, for comparison only)

Airfoil		Cd min R=6M	Cm c/4 R=6M	Cl max-no flaps R=2M	Cl max-w/flaps* R=2M
GA30 -613.5		.0068	-.105	1.789	3.62
NACA 4415	(1931)	.0073	-.126	1.709	3.37
NACA 4412	(1931)	.0070	-.126	1.707	3.14
USA 35B	(1923)	.0071	-.114	1.677	3.02
Clark Y	(1924)	.0072	-.111	1.622	3.07

*Flap condition: Cflap=.20, deflection=25 degrees

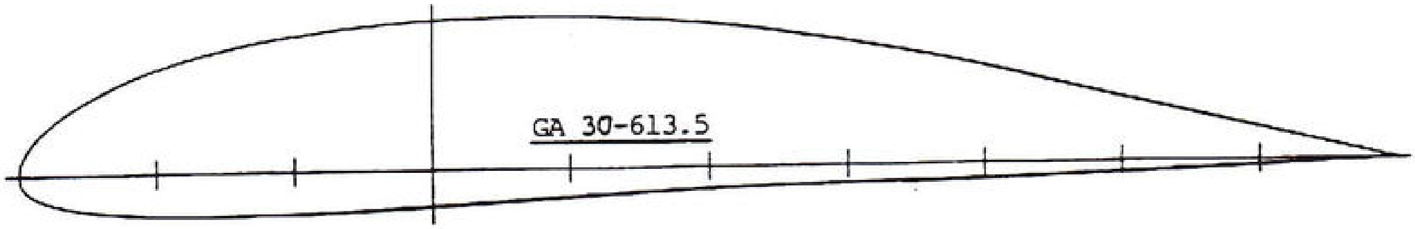


FIGURE III-19

CLASSIC HIGH-LIFT AIRFOILS (FOR COMPARISON ONLY)

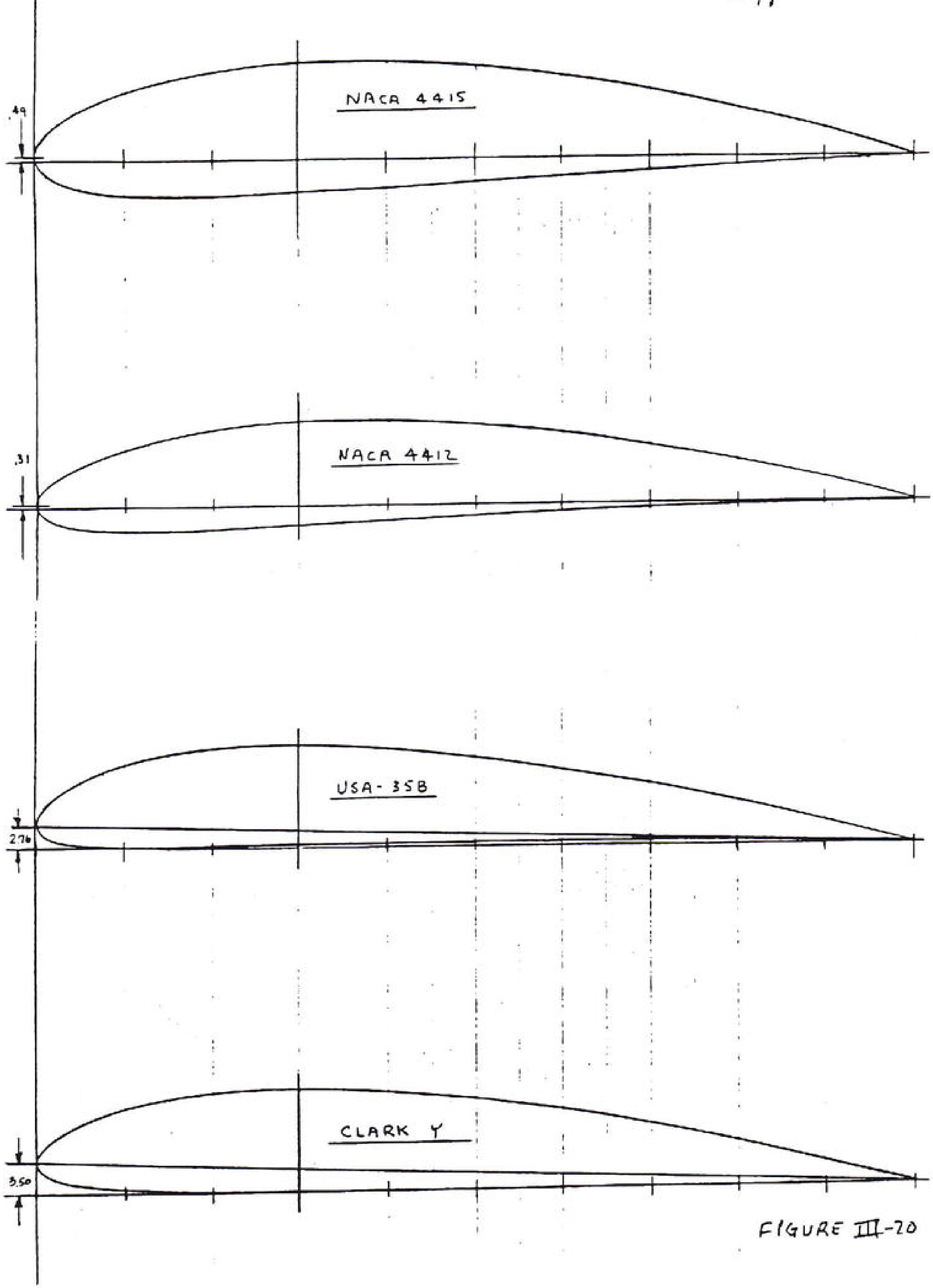

.49
NACA 4415
.31
NACA 4412
USA-35B
2.76
CLARK Y
3.50

FIGURE III-20

Figure IV - Predicted Performance Curves, from computer analysis. (12% and 15% thick shapes)

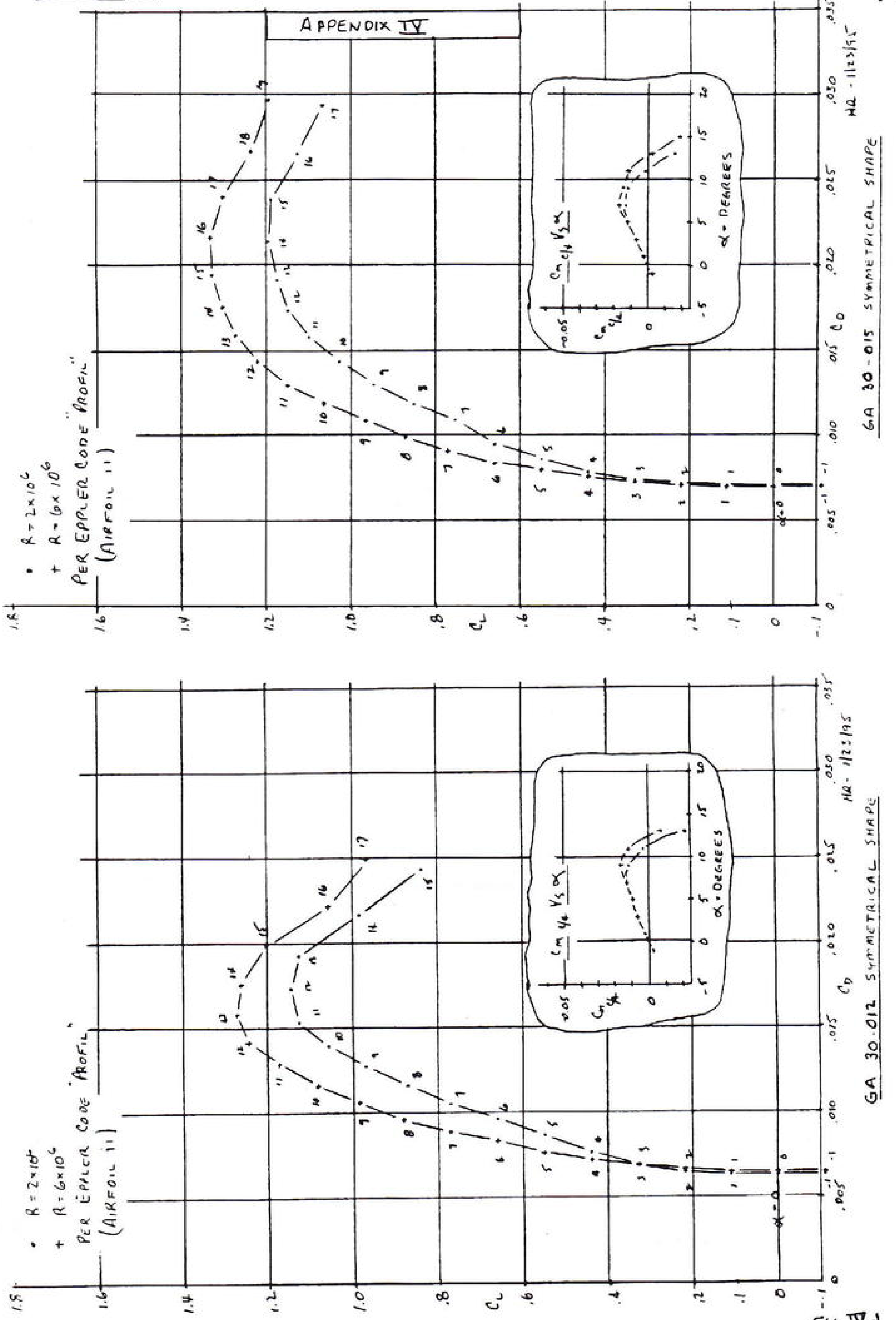
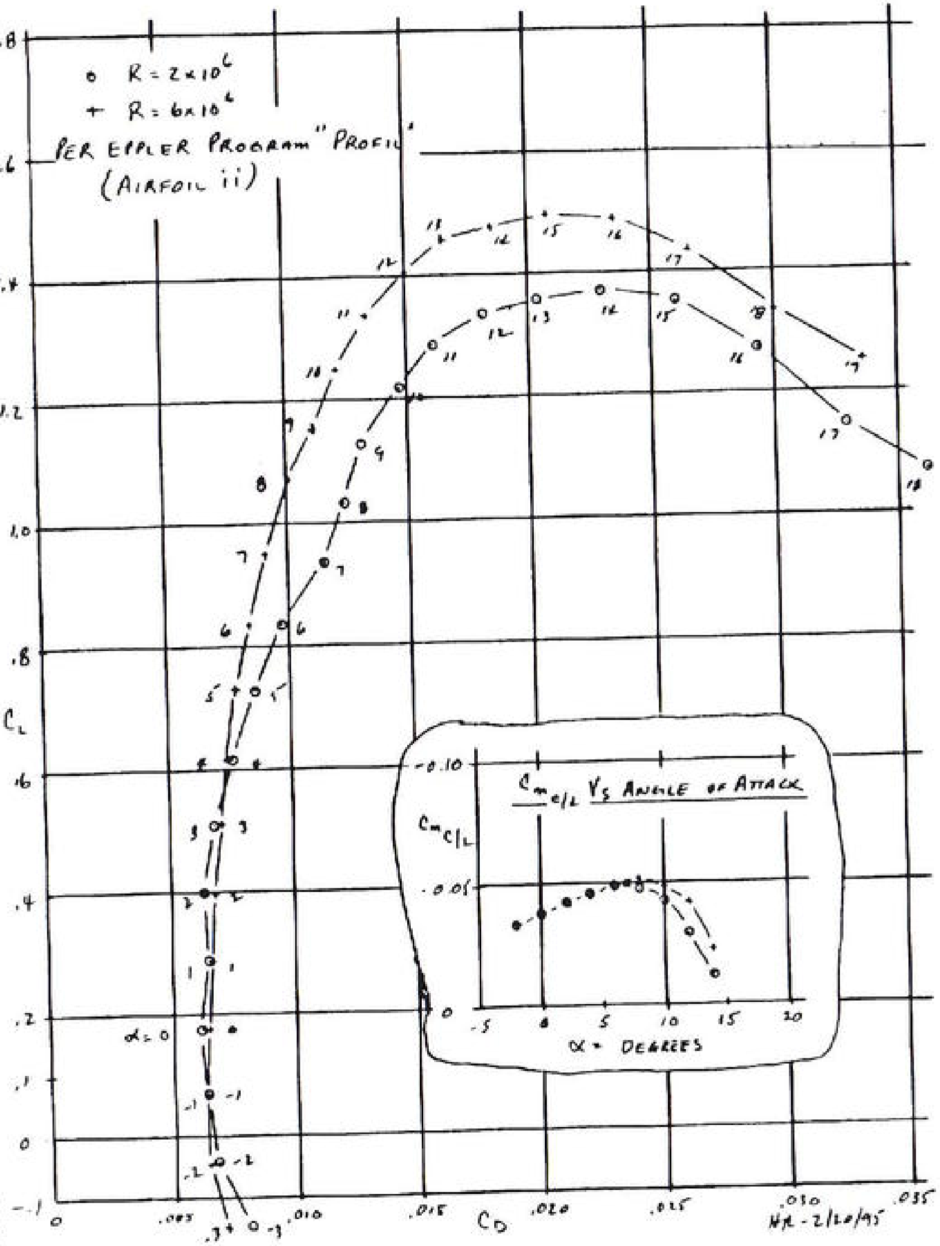
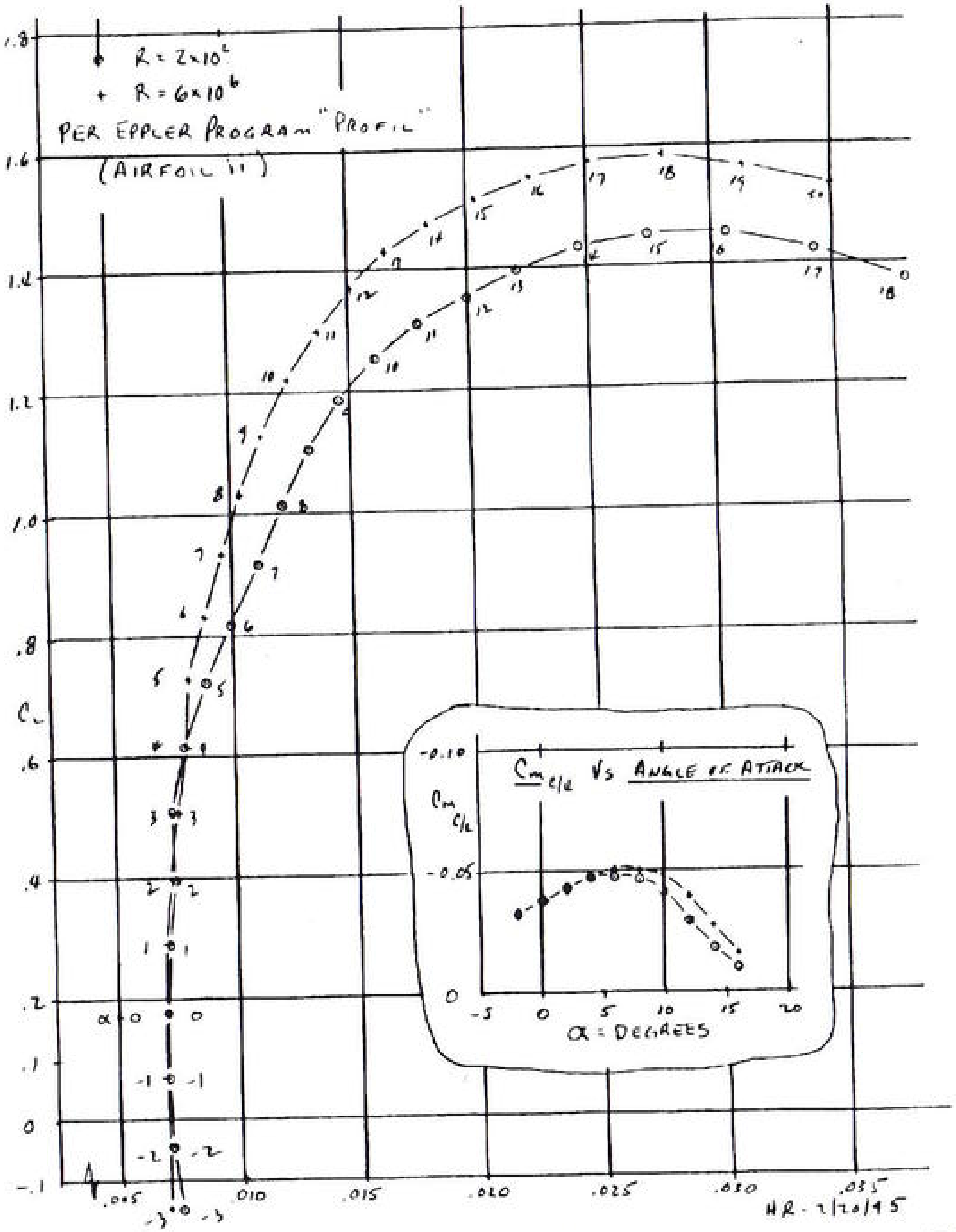


FIG IV-1

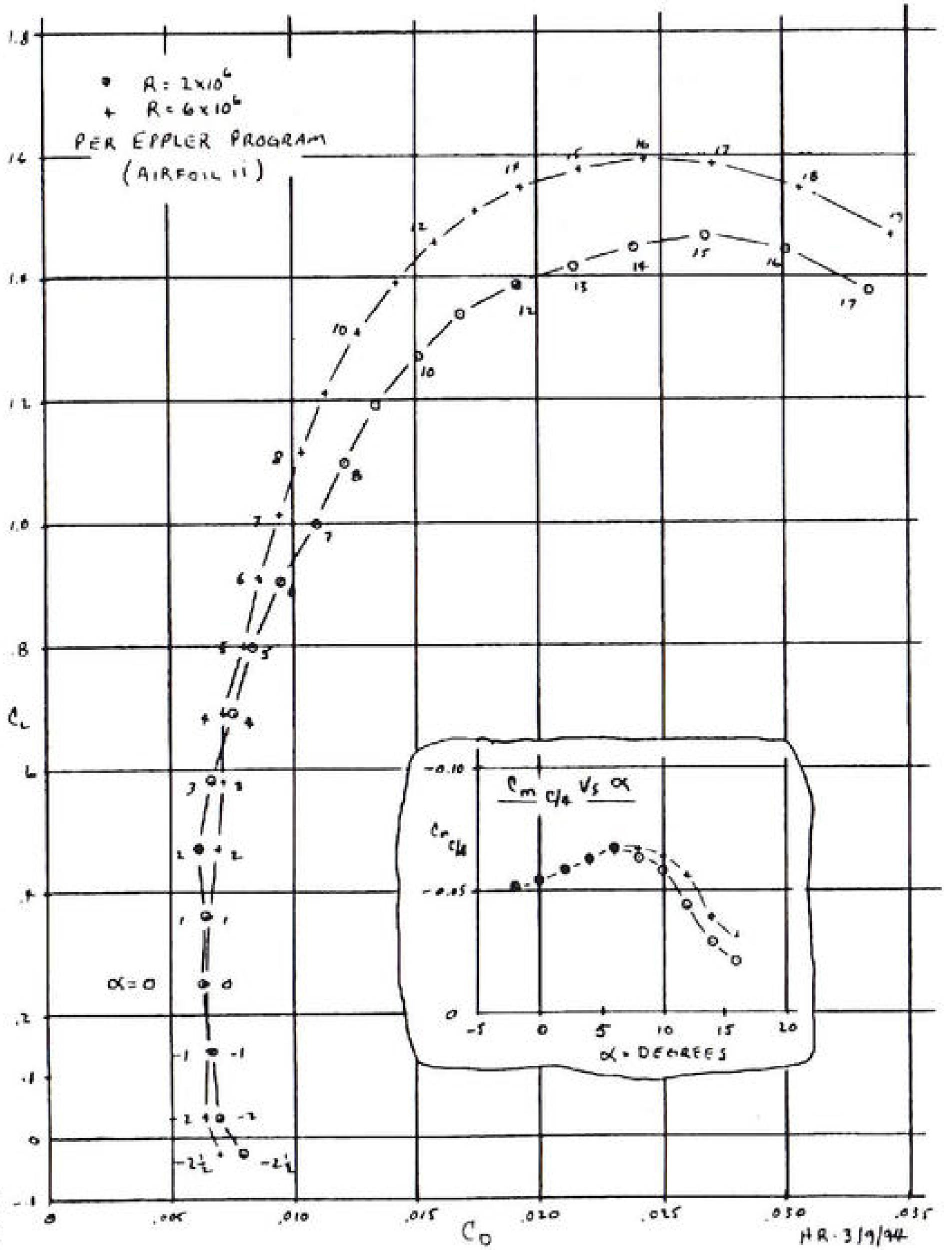


GA 30-212 AIRFOIL

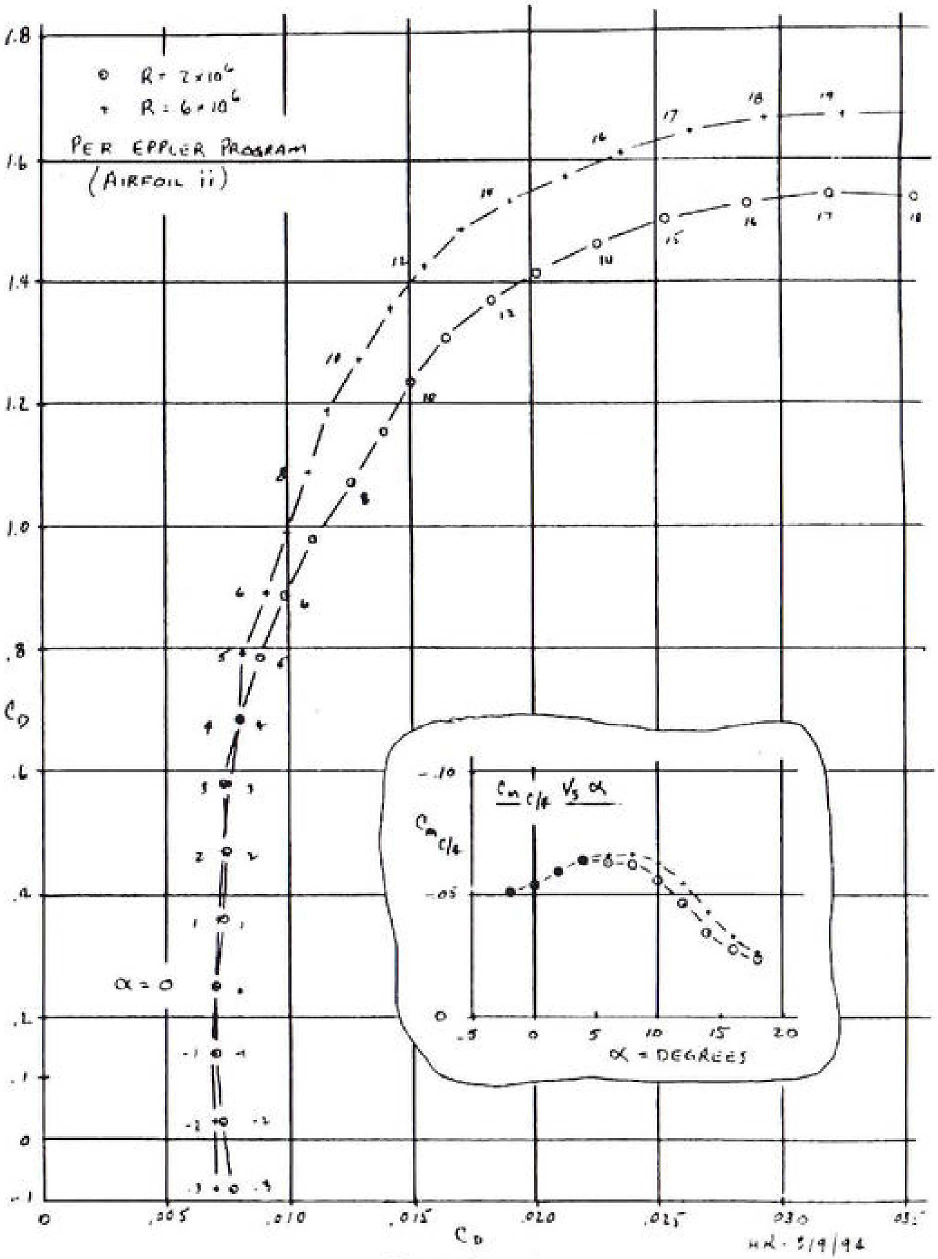


GA 30-215 AIRFOIL

FIG IV-2

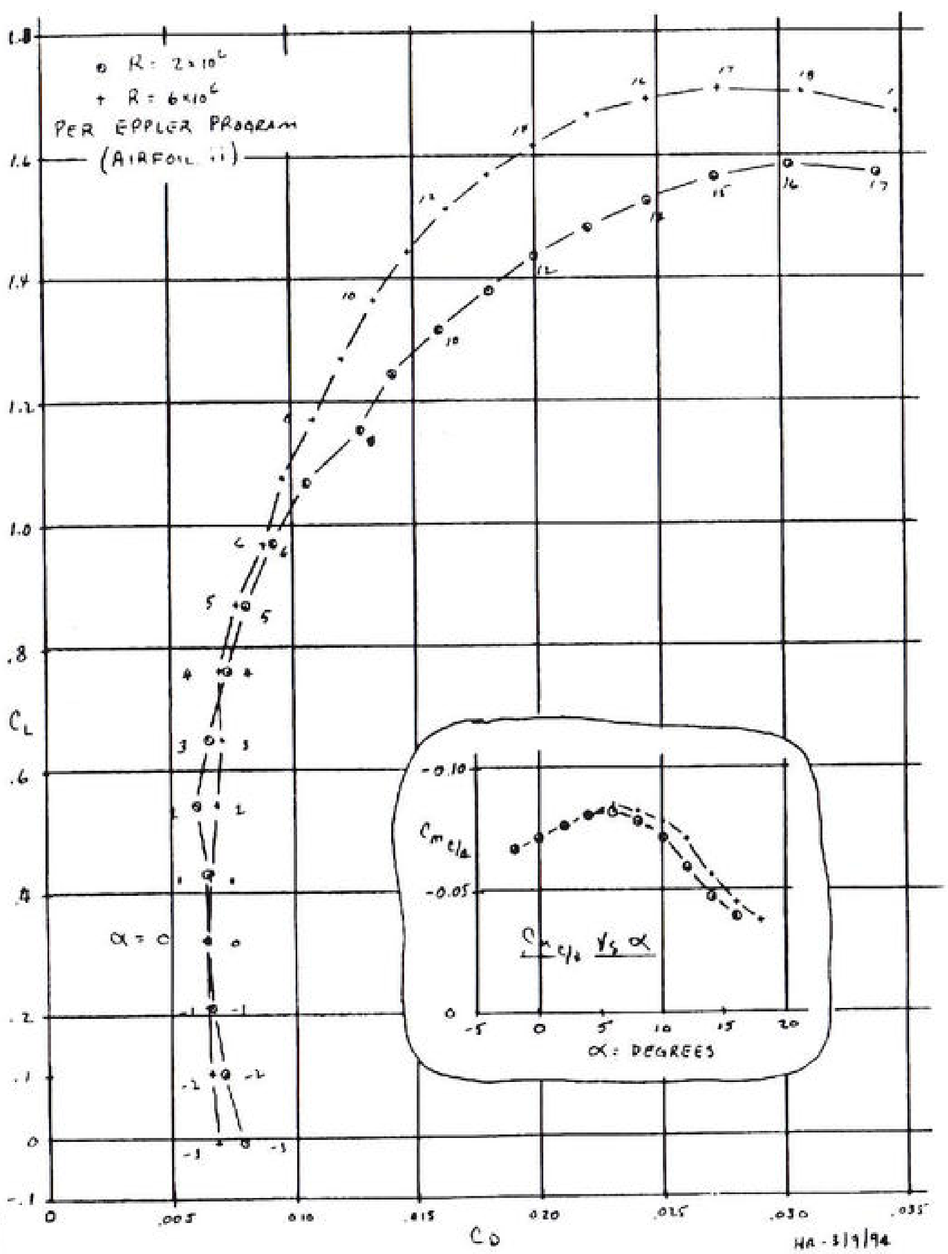


GA 30-312 AIRFOIL



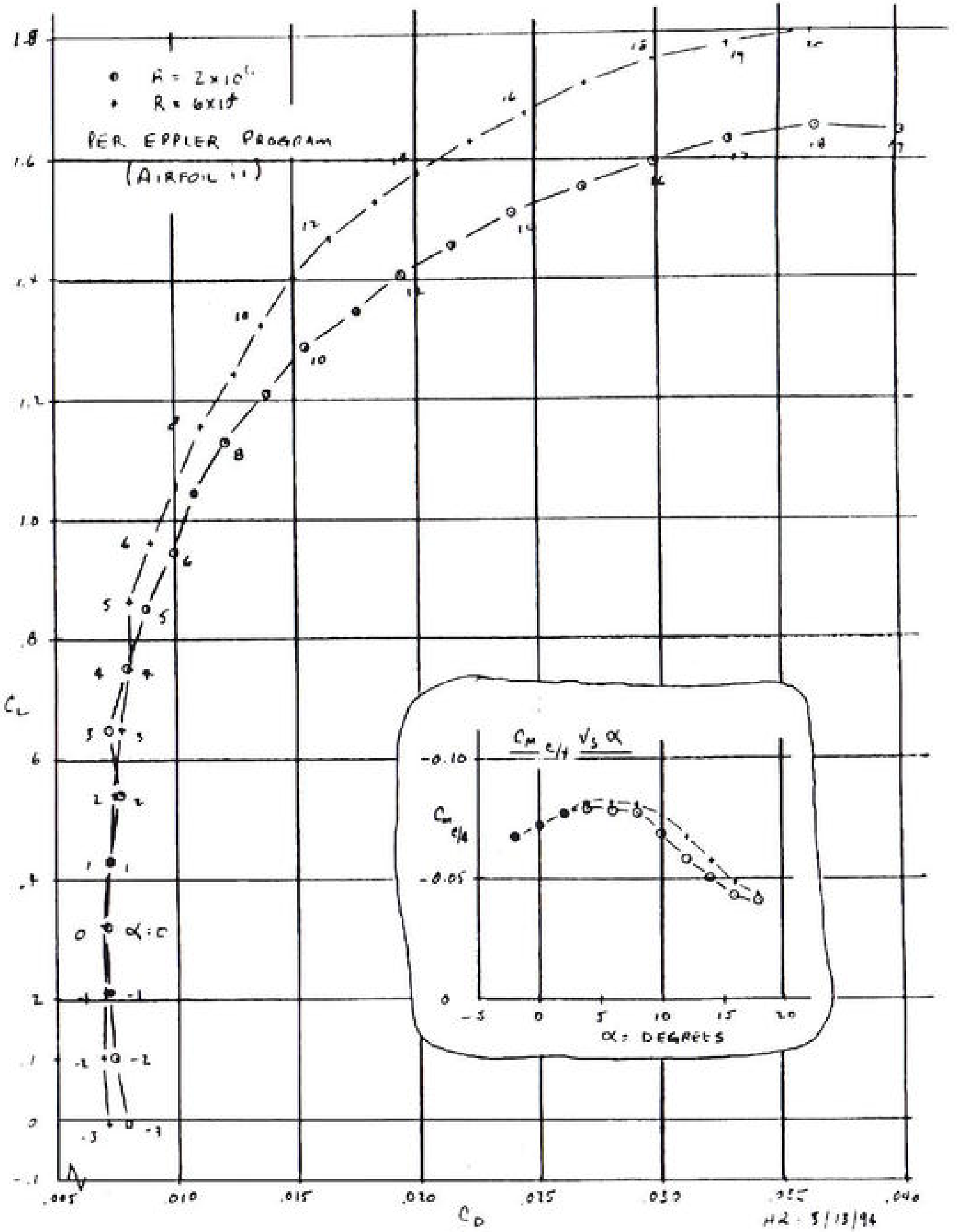
GA 30-315 AIRFOIL

FIG III-3



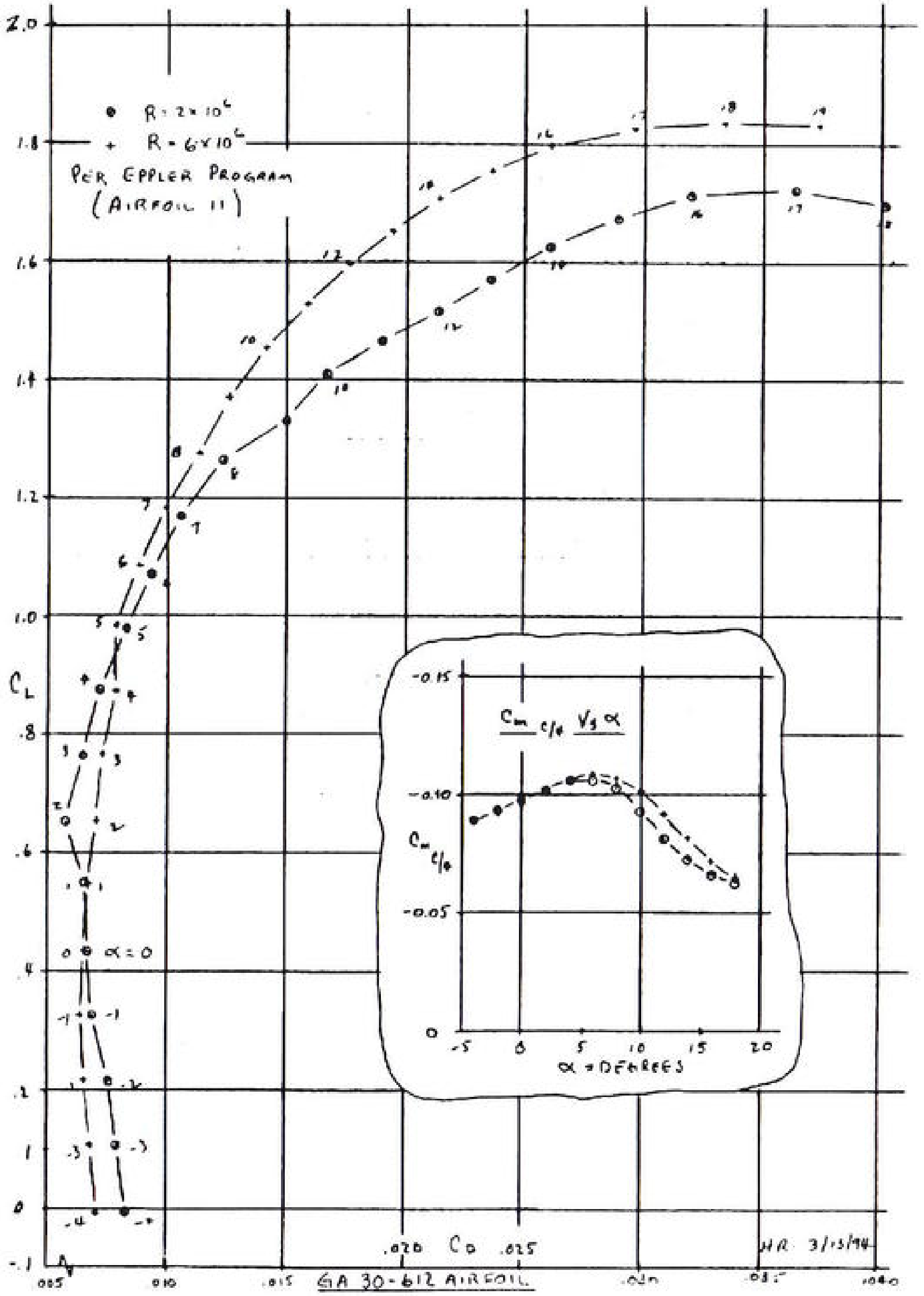
HA - 3/19/94

GA 30-412 AIRFOIL

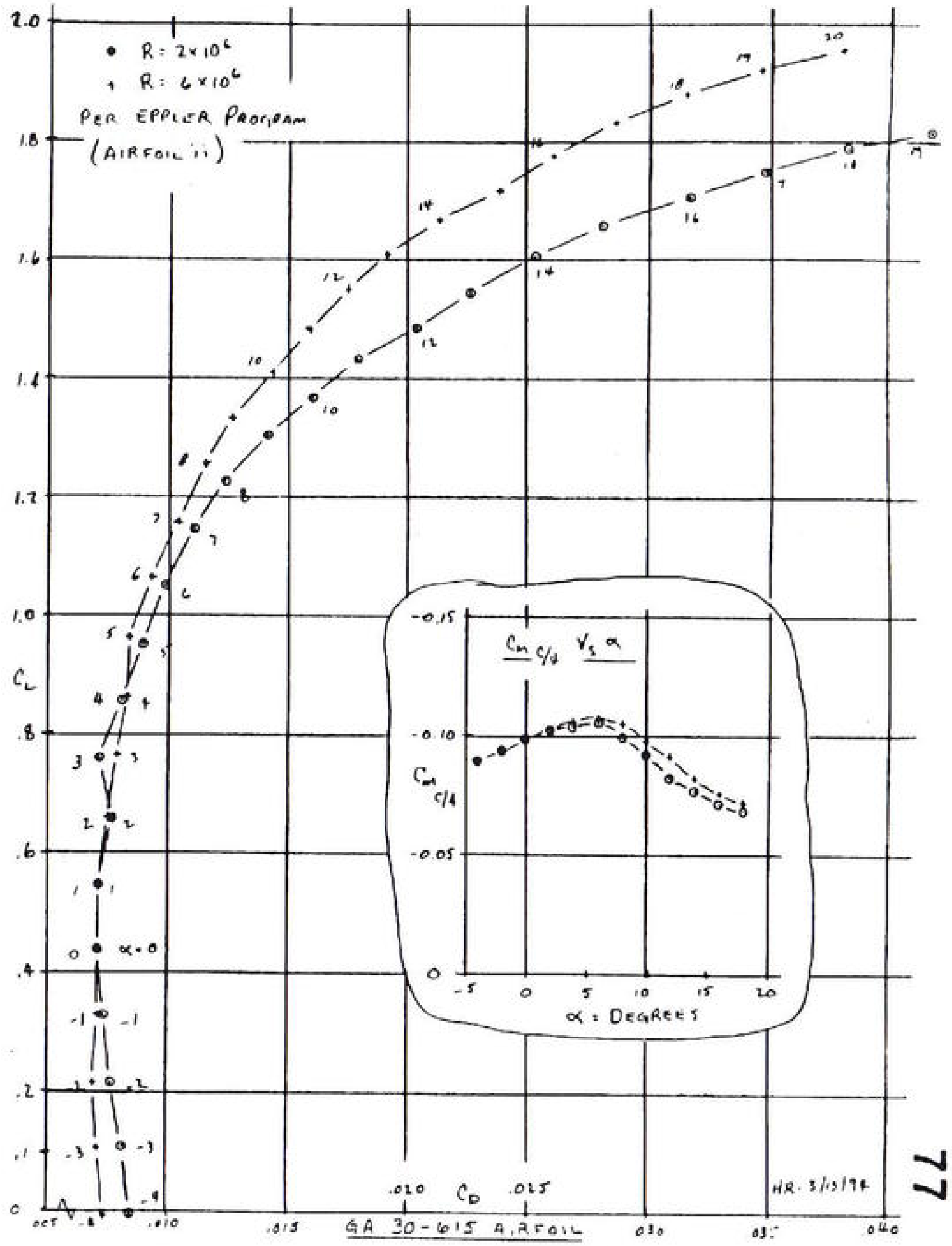


HA - 3/13/94

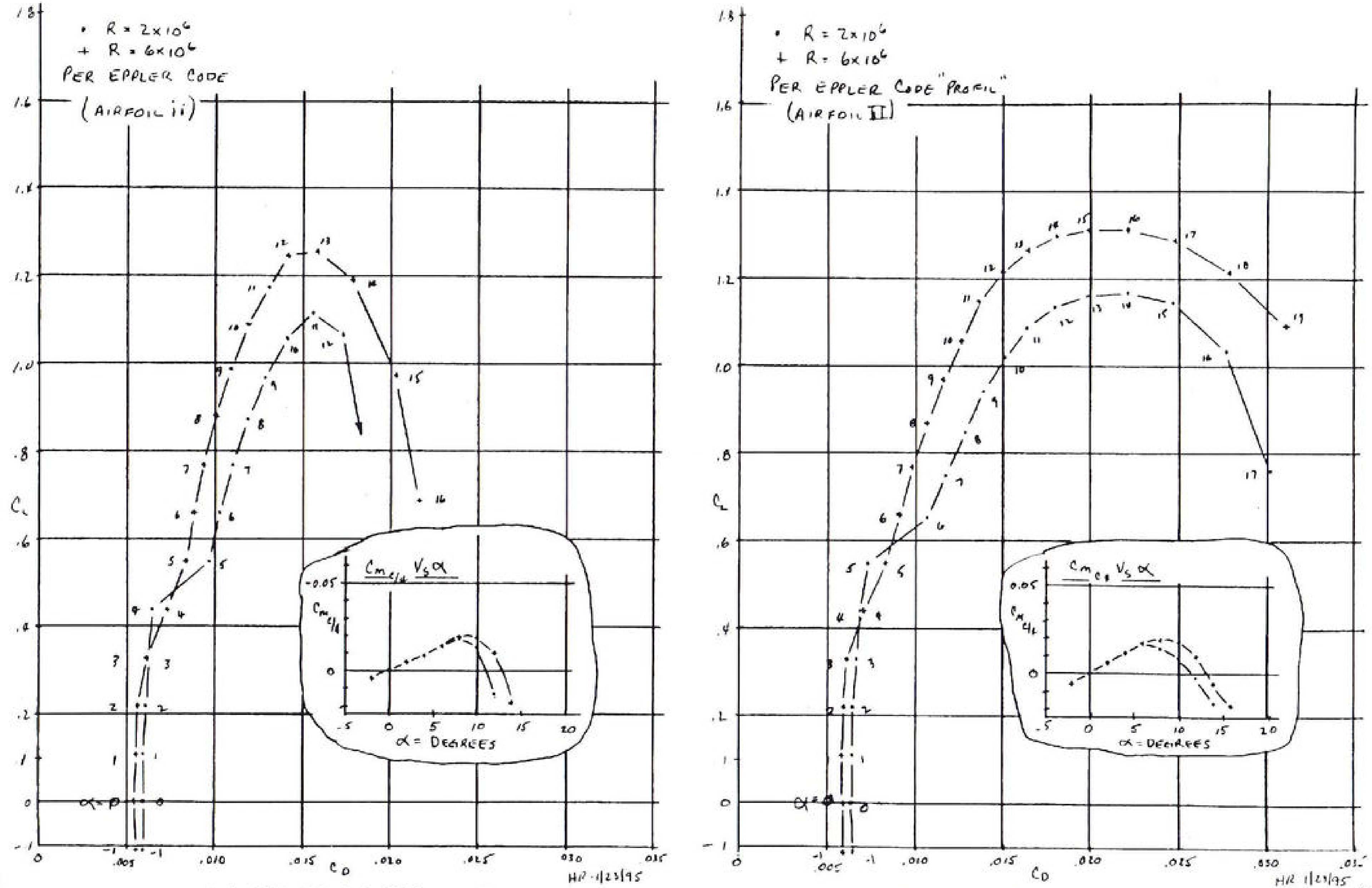
GA 30-415 AIRFOIL



GA 30-612 AIRFOIL



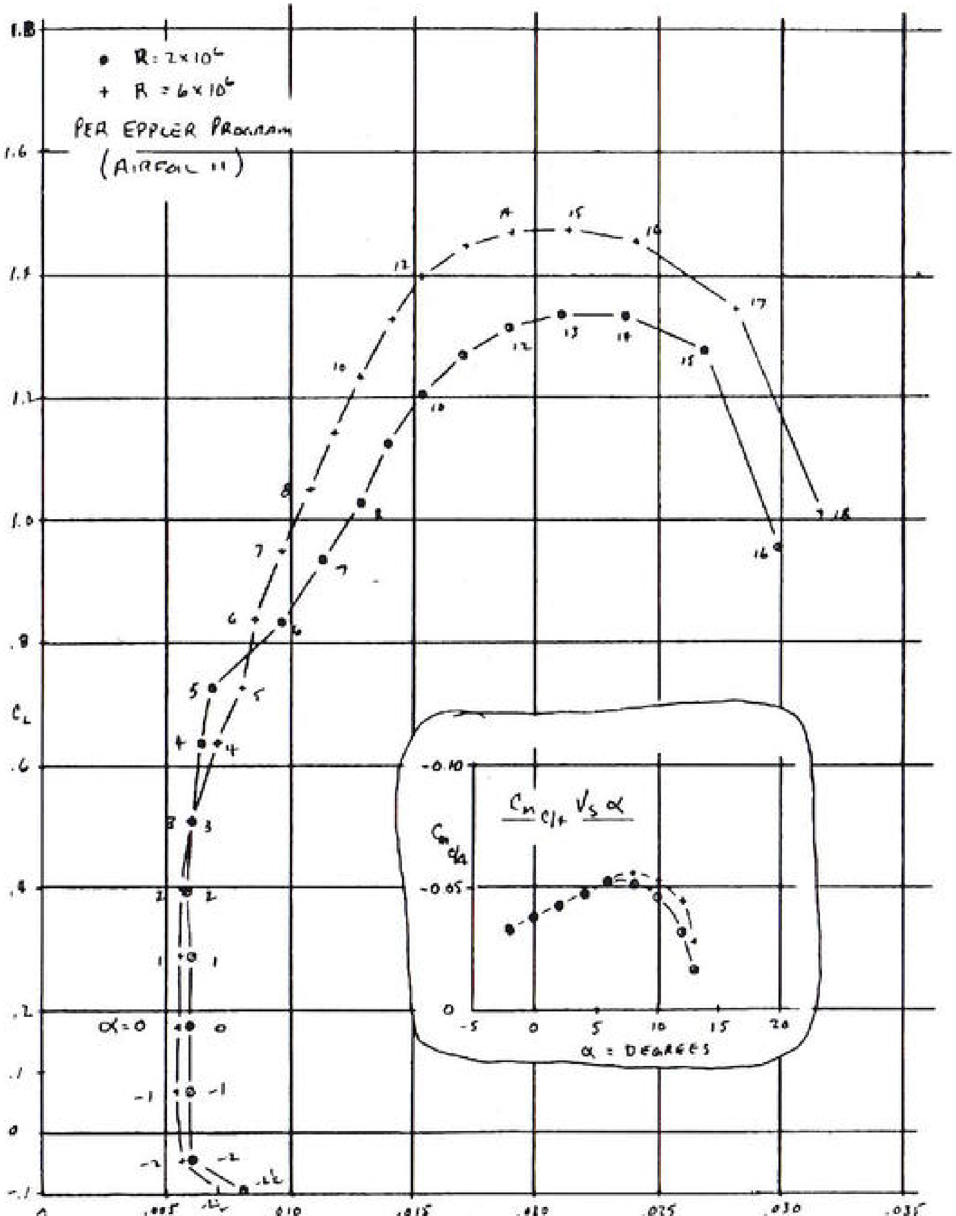
GA 30-615 AIRFOIL



GA 35A012 SYMMETRICAL SHAPE

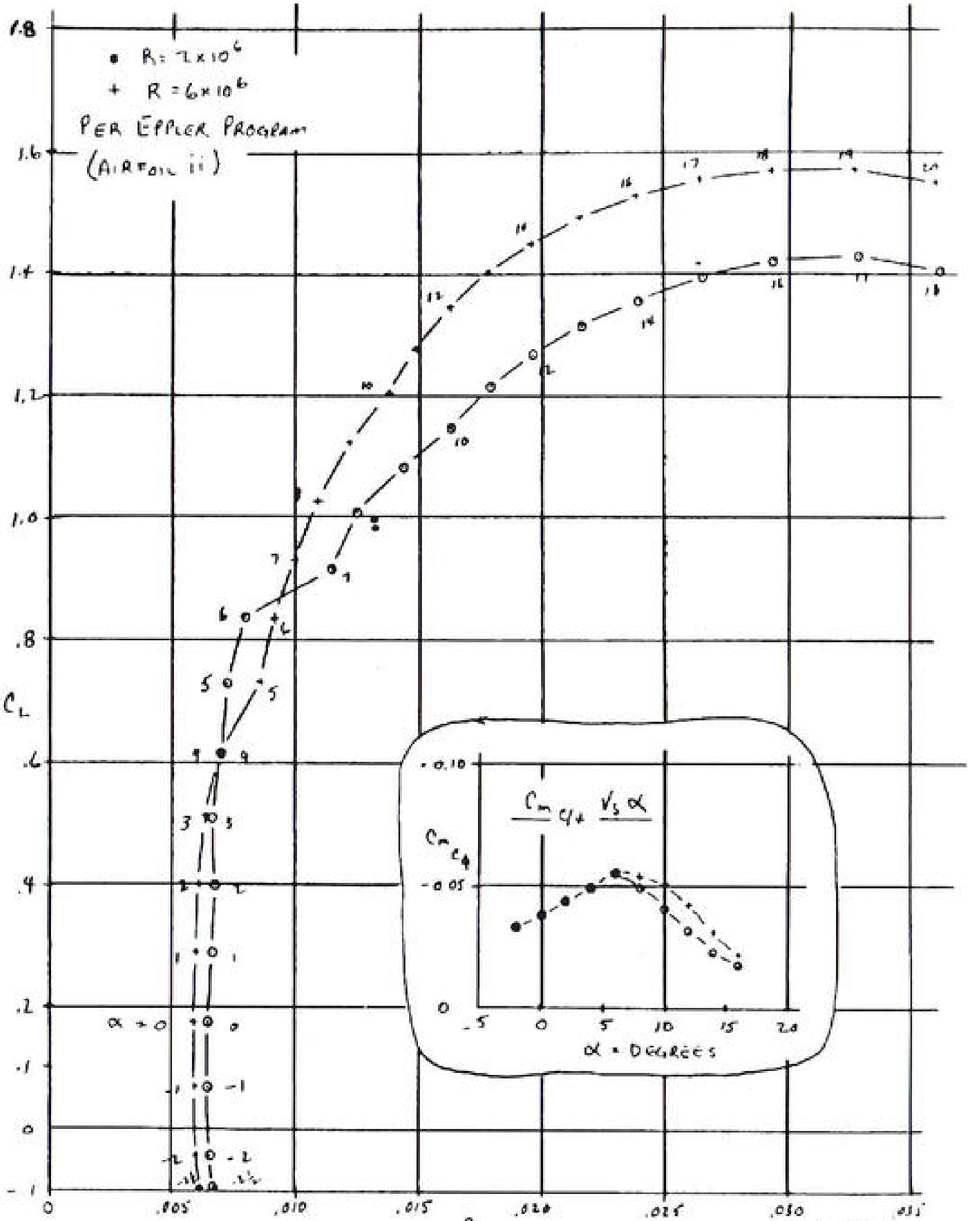
GA 35A015 SYMMETRICAL SHAPE (NACA 63A015)

FIG IV-6



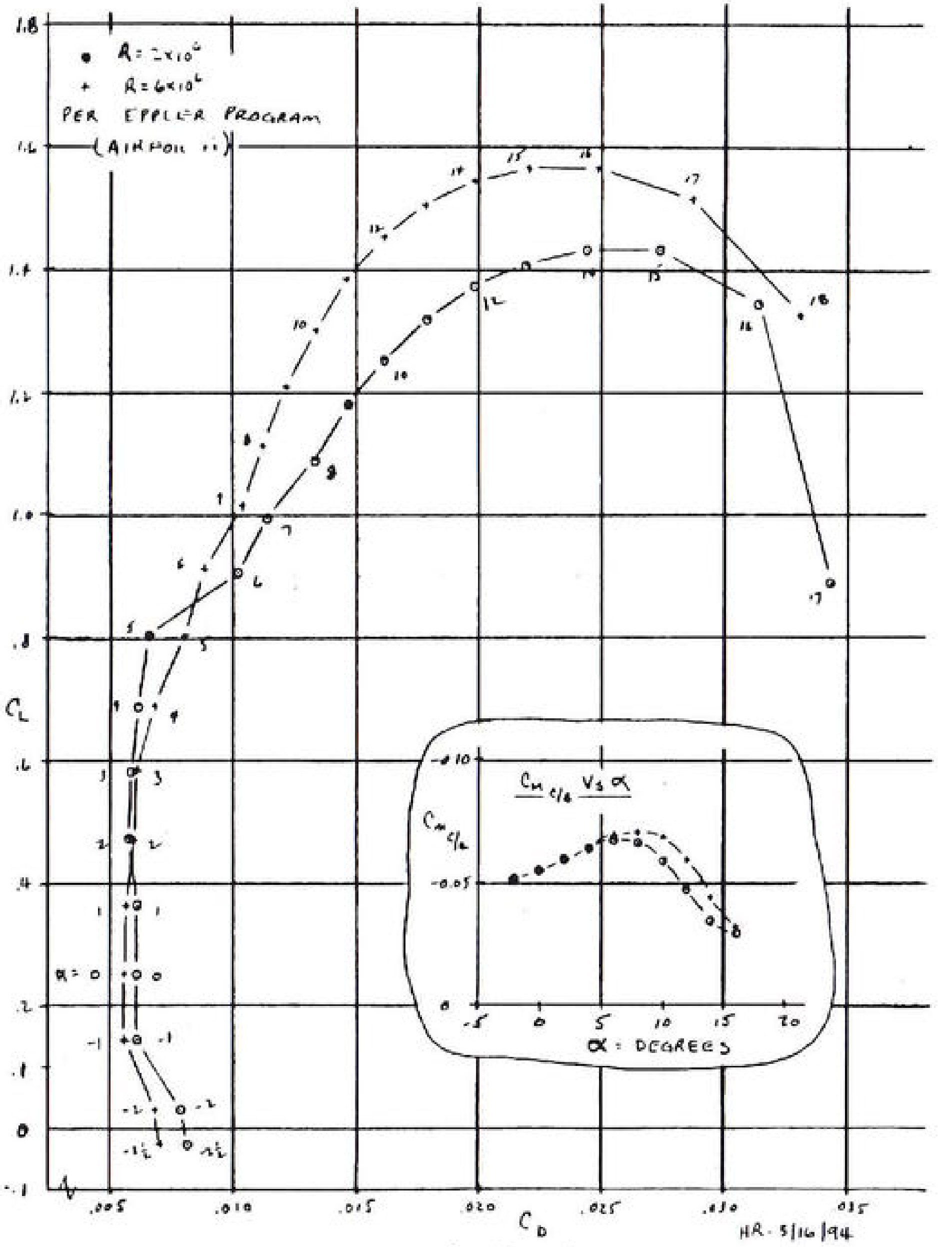
HR-3/15/94

GA 35A212 AIRFOIL

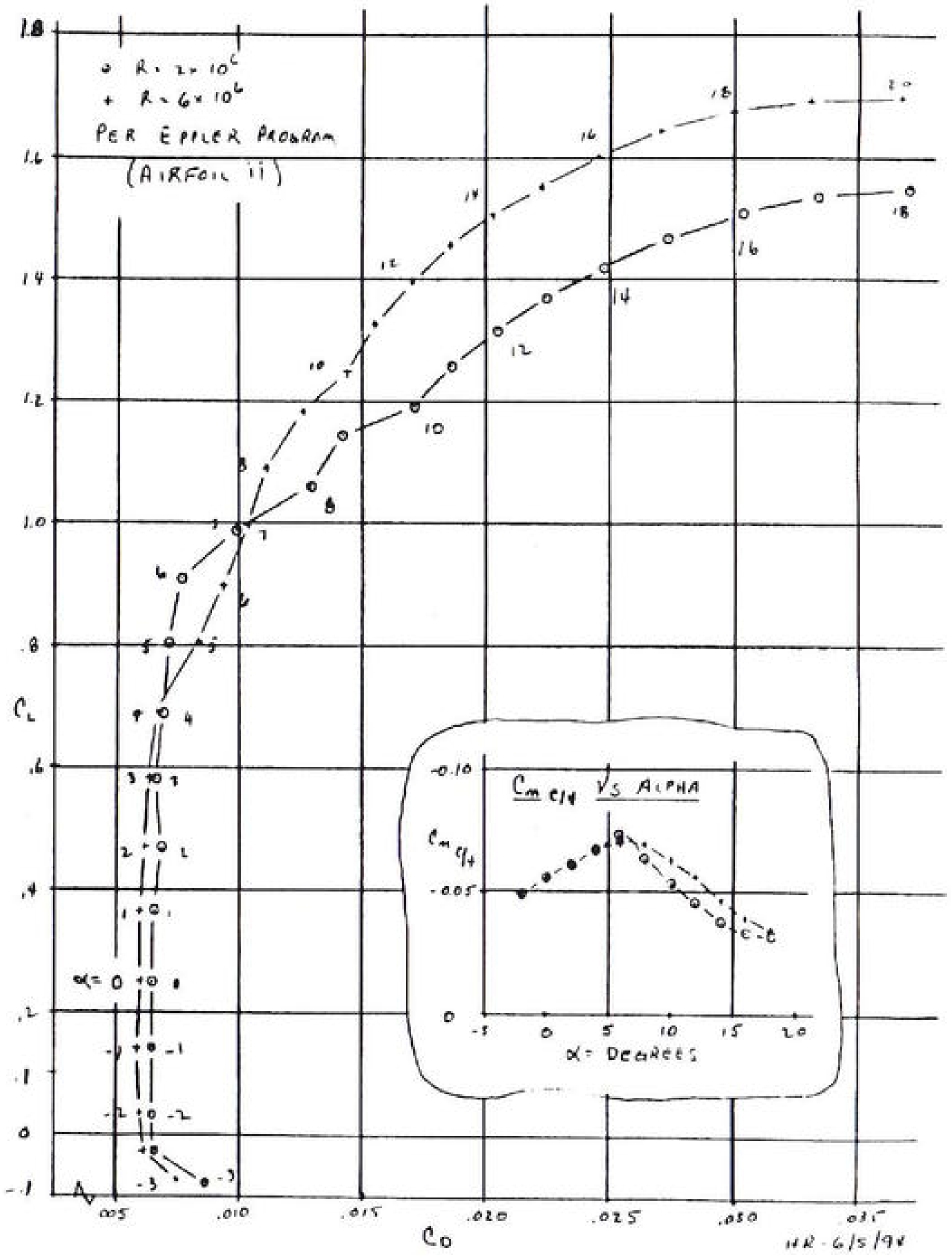


HR-3/15/94

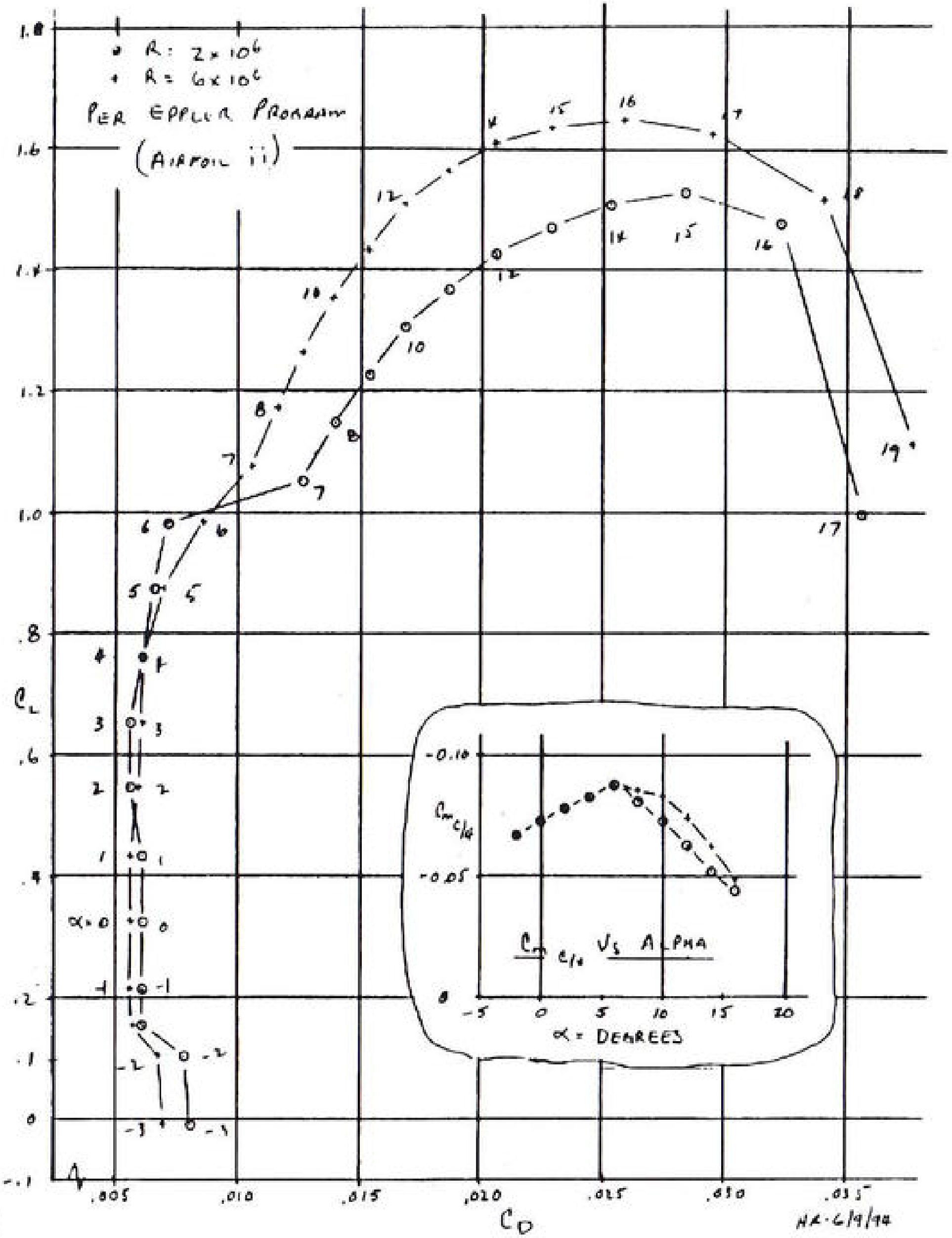
GA 35A215 AIRFOIL



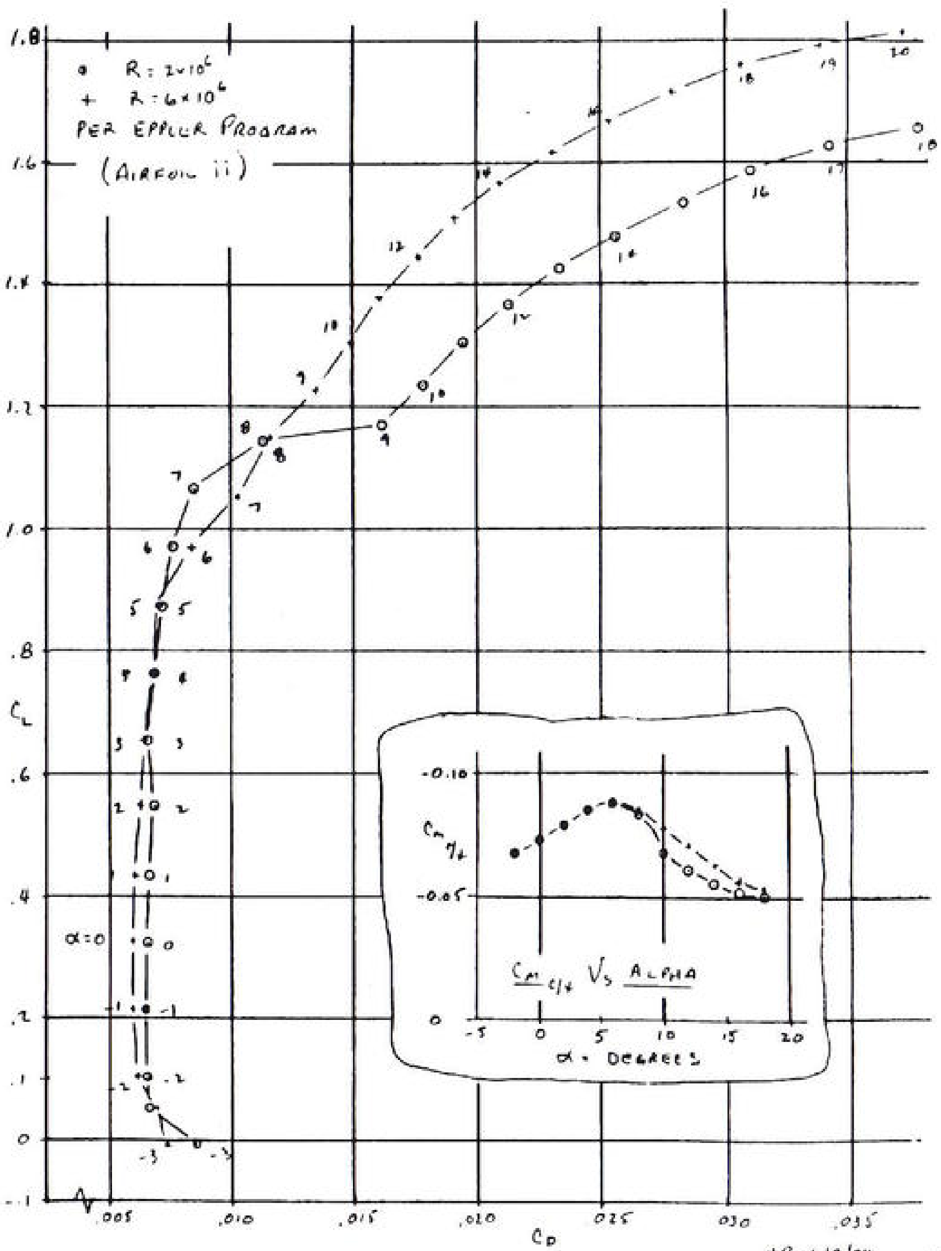
GA 35A312 AIRFOIL



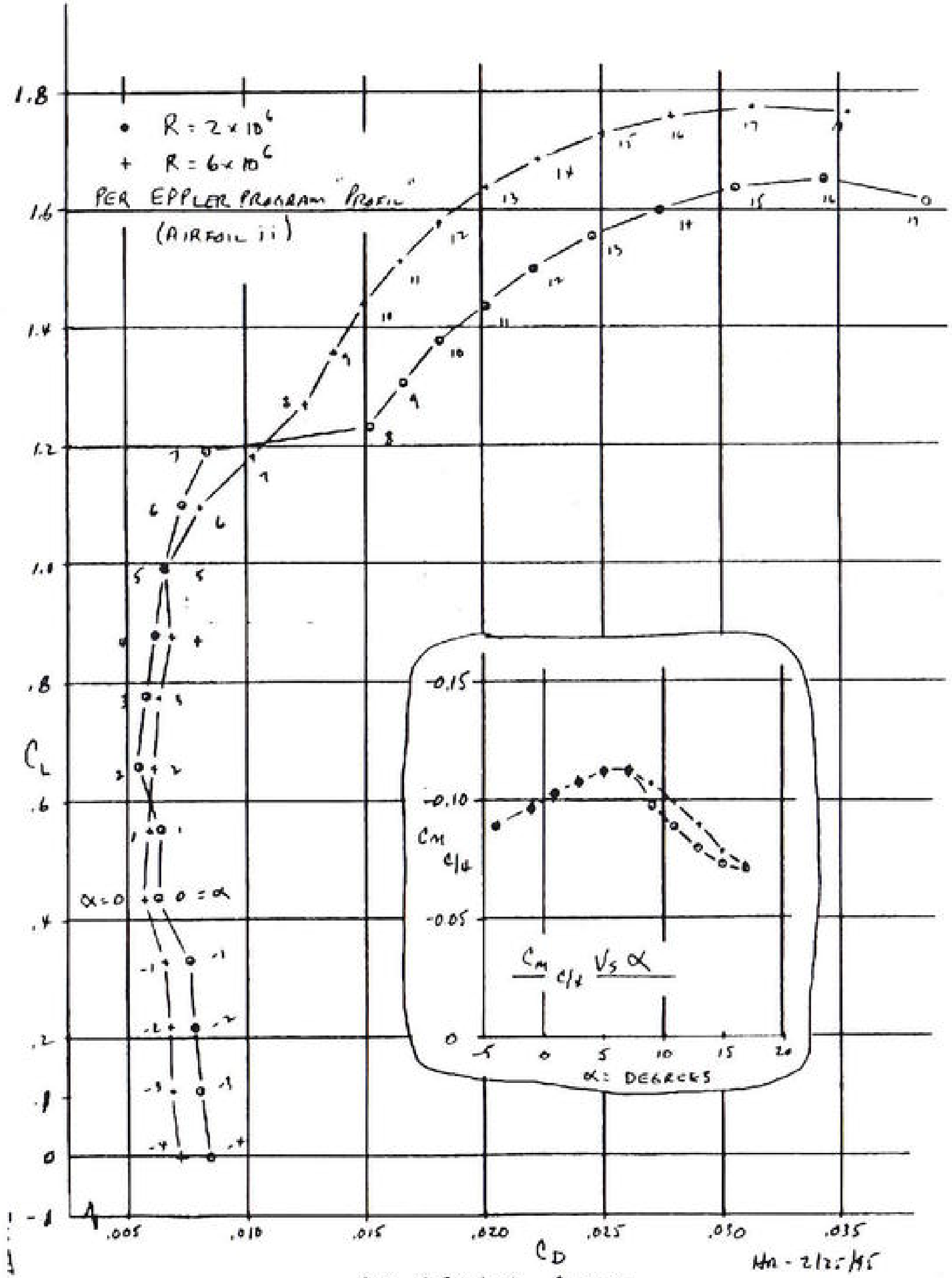
GA 35A315 AIRFO.



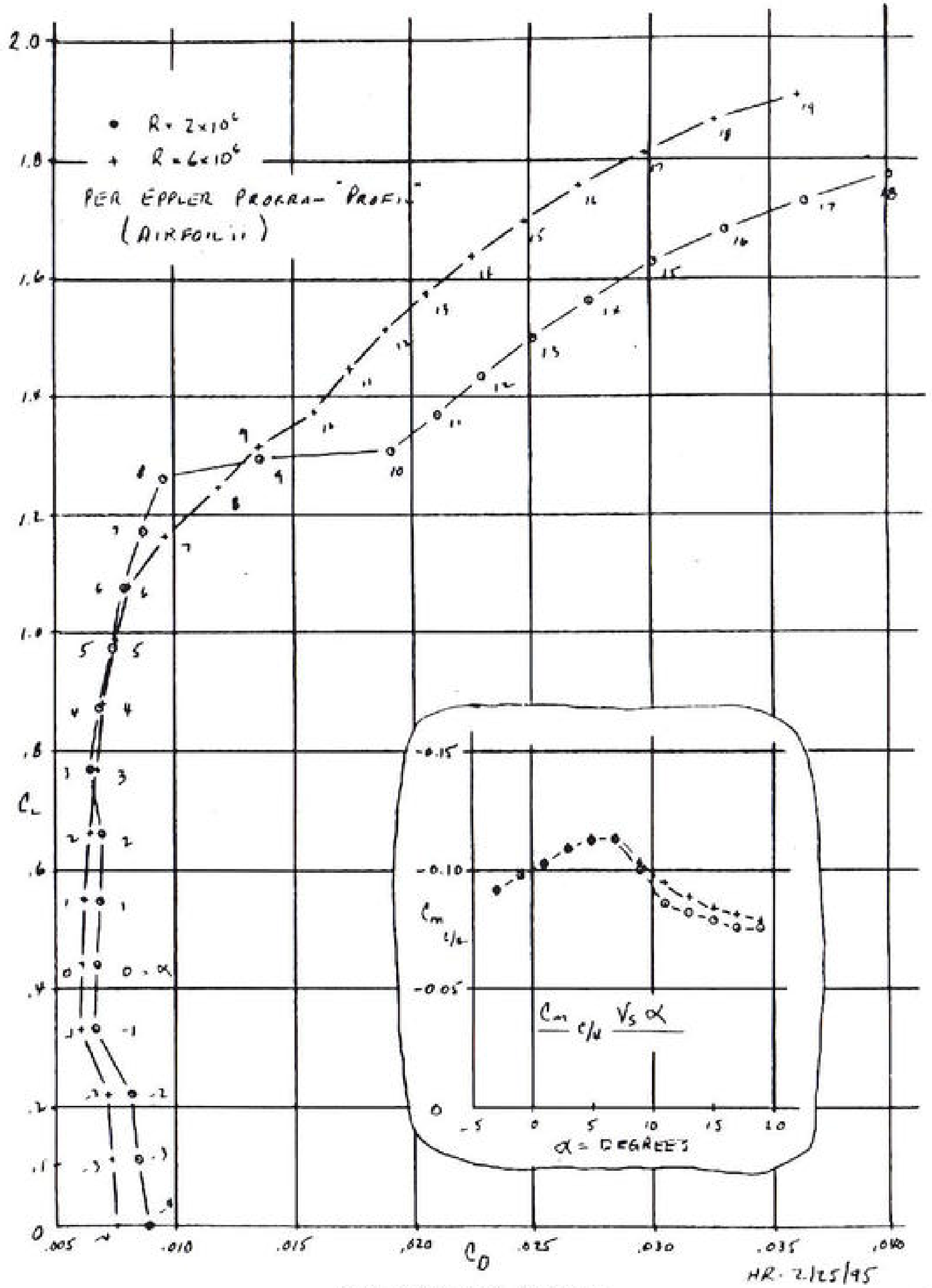
GA 35A412 AIRFOIL



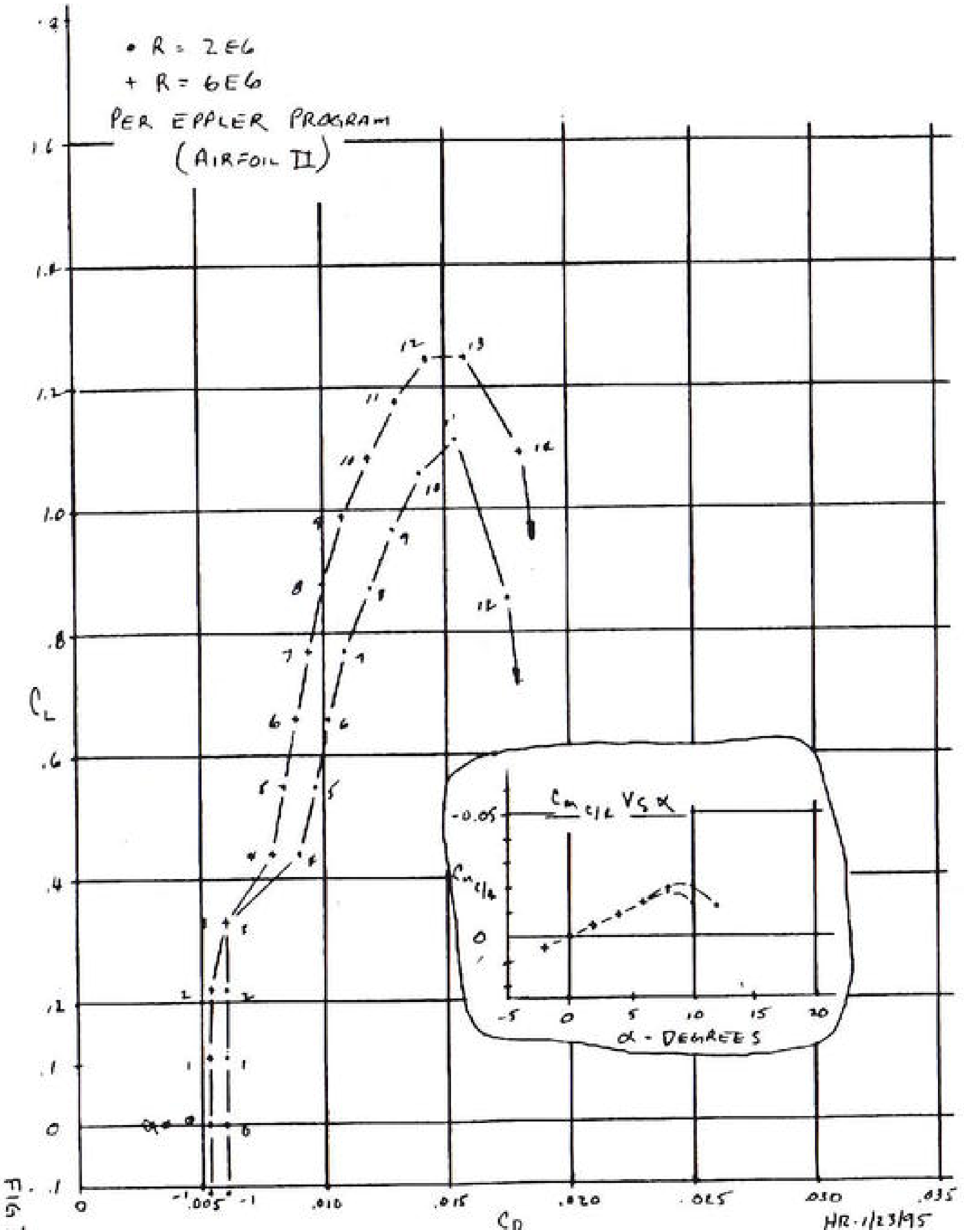
GA 35A415 AIRFOIL



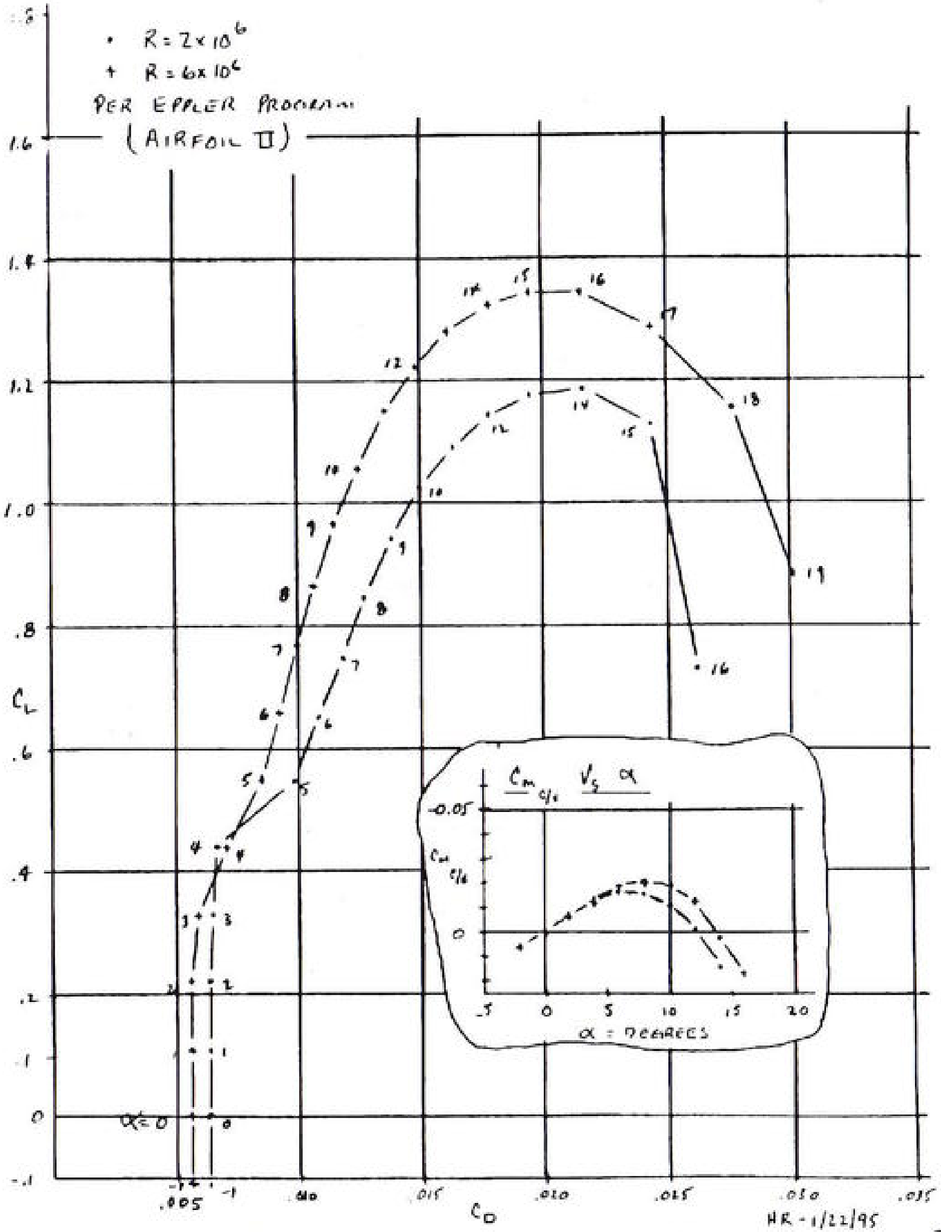
GA 35A612 AIRFOIL



GA 35A615 AIRFOIL

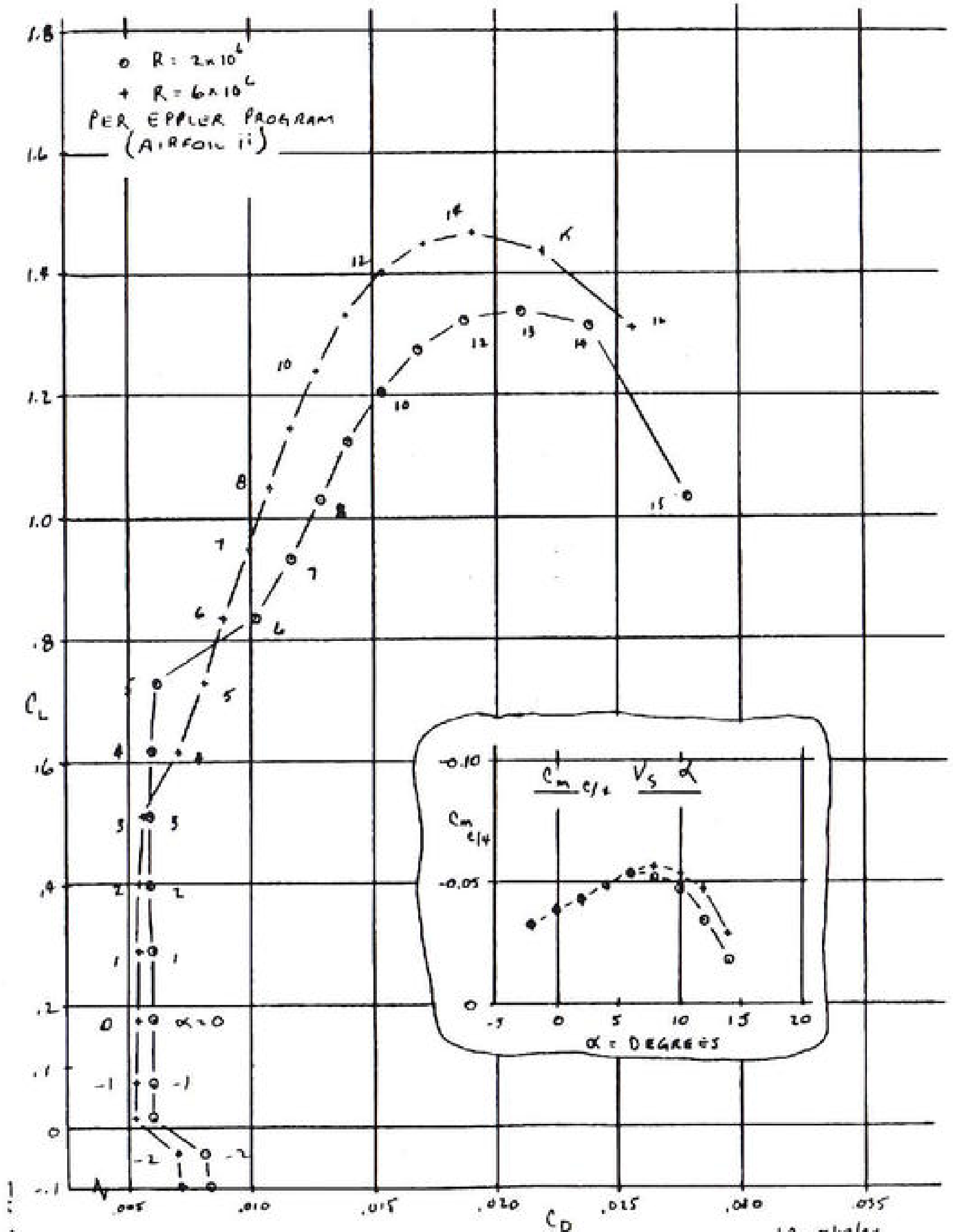


GA 37A012 SYMMETRICAL SHAPE



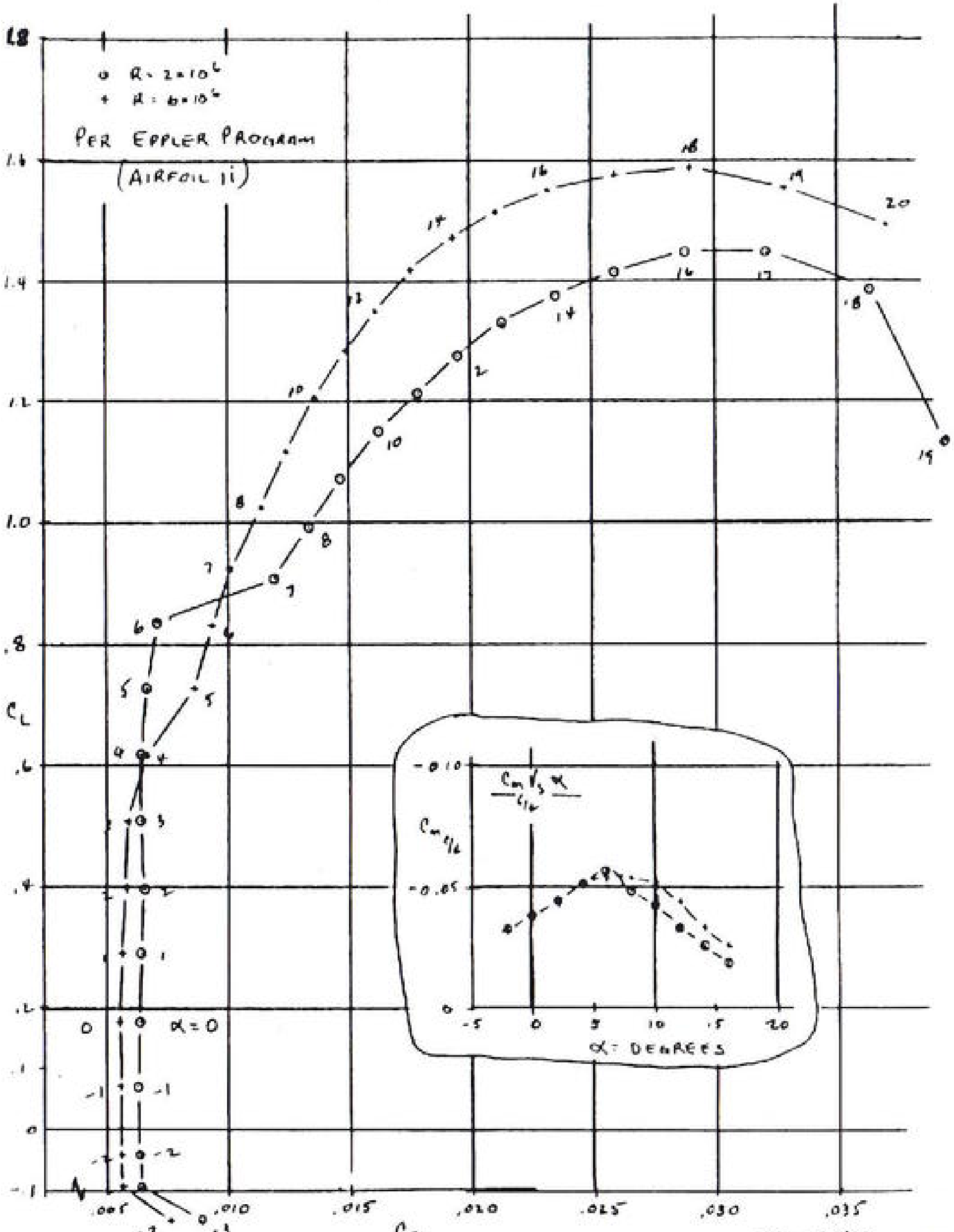
GA 37A015 SYMMETRICAL SHAPE (NACA 64A015)

FIG IV-11



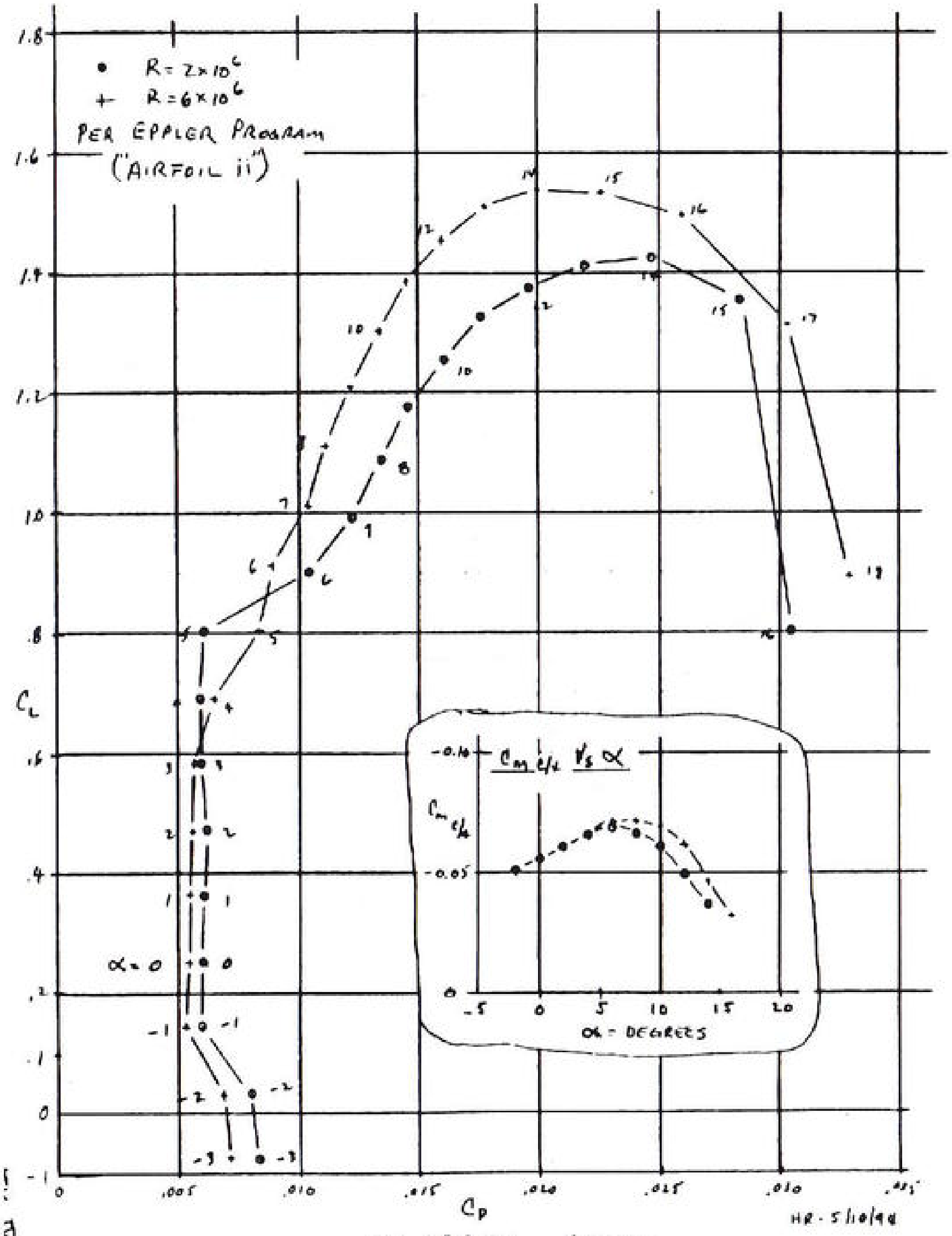
HR - 5/19/94

GA 37 A 212 AIRFOIL

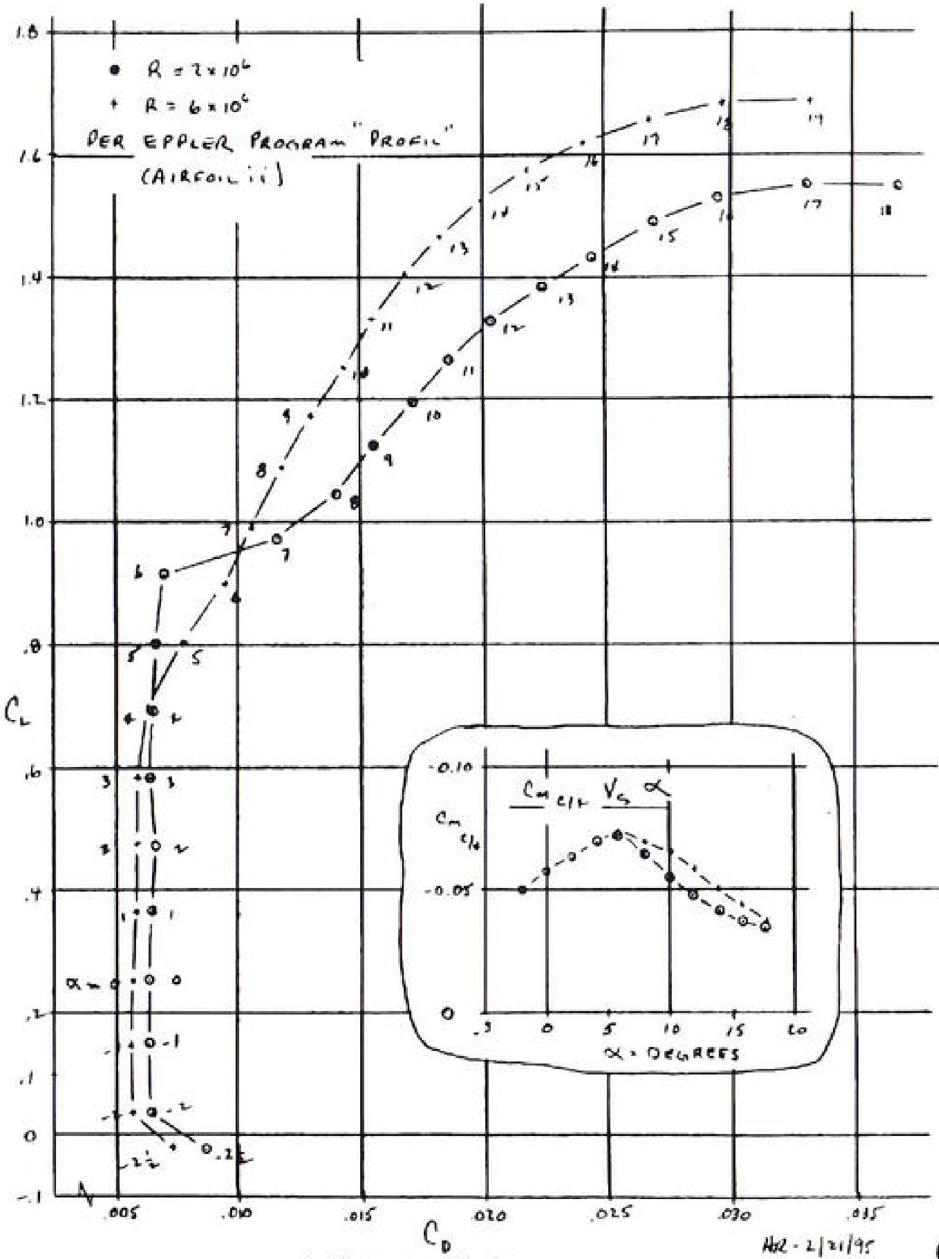


HR - 5/13/94

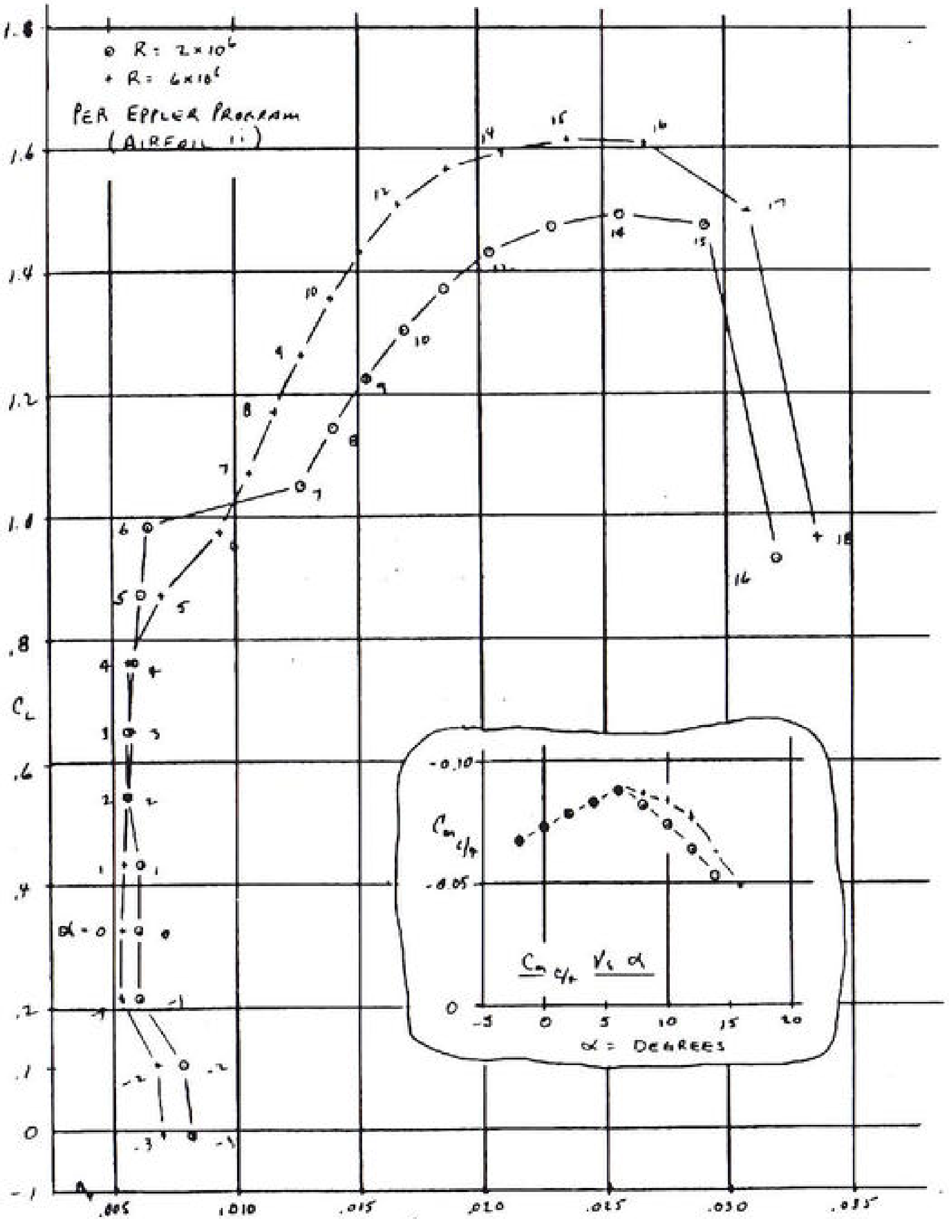
GA 37 A 215 AIRFOIL



GA 37A312 AIRFOIL

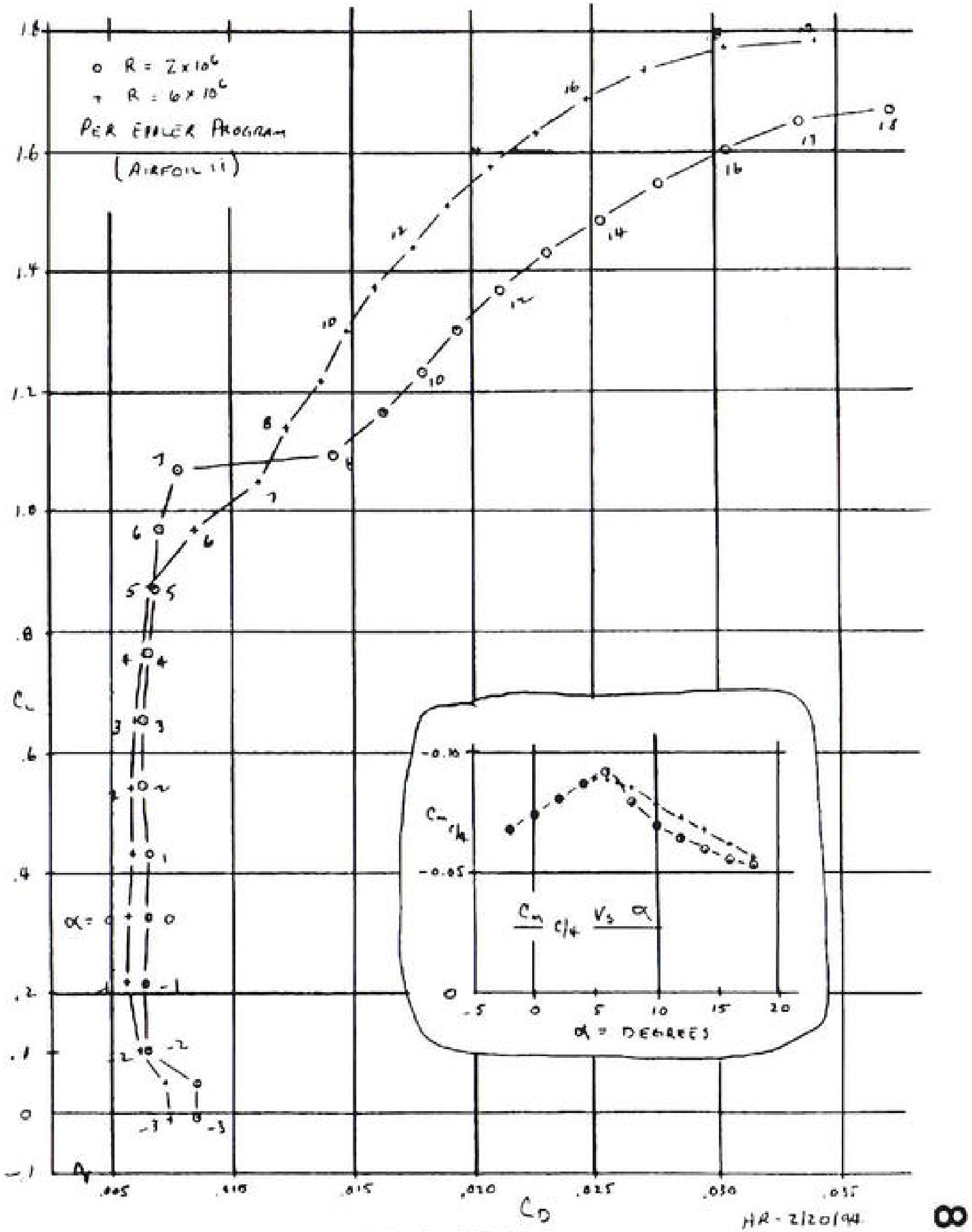


GA 37A315 AIRFOIL



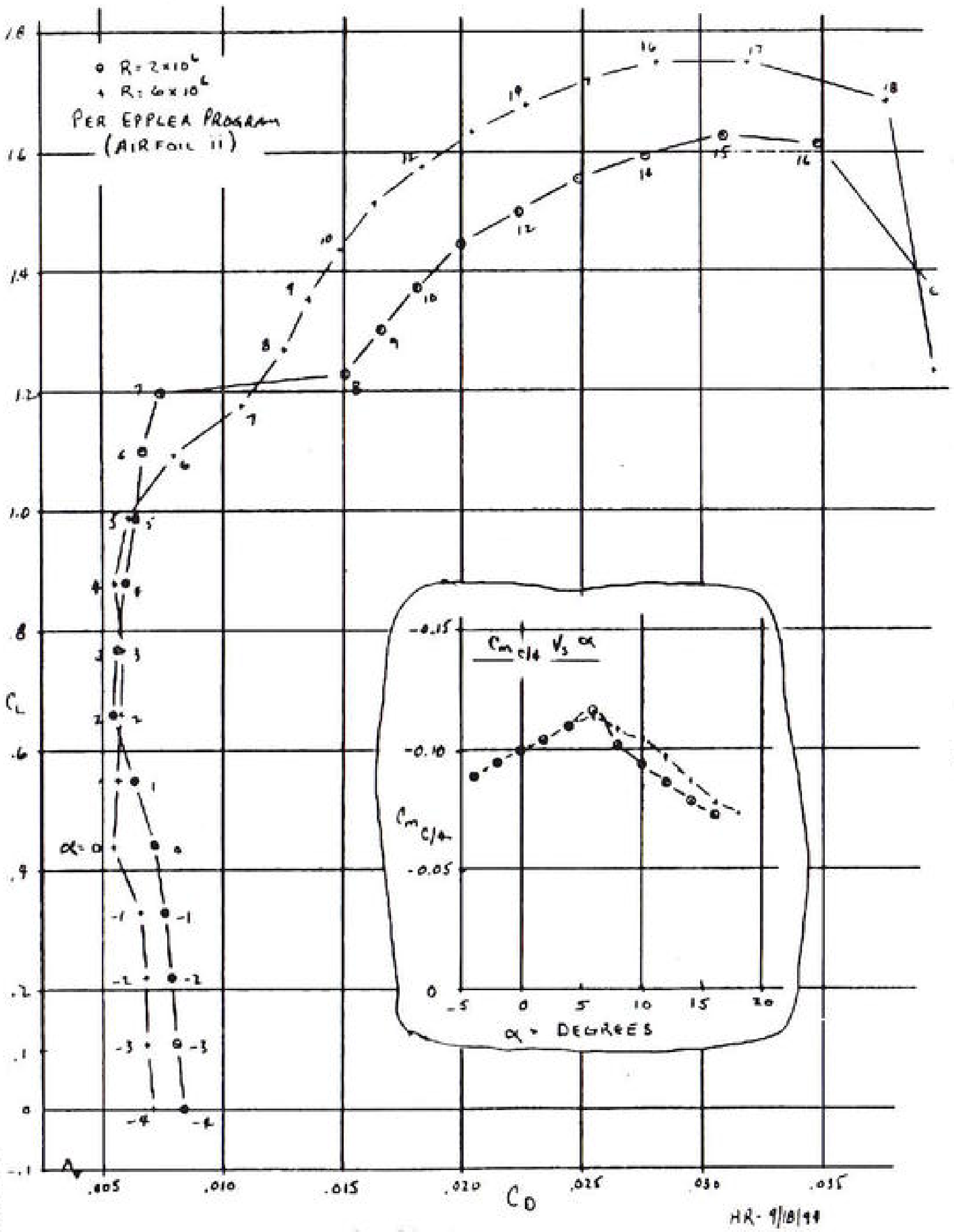
HR - 5/19/94

GA 37A 412 AIRFOIL

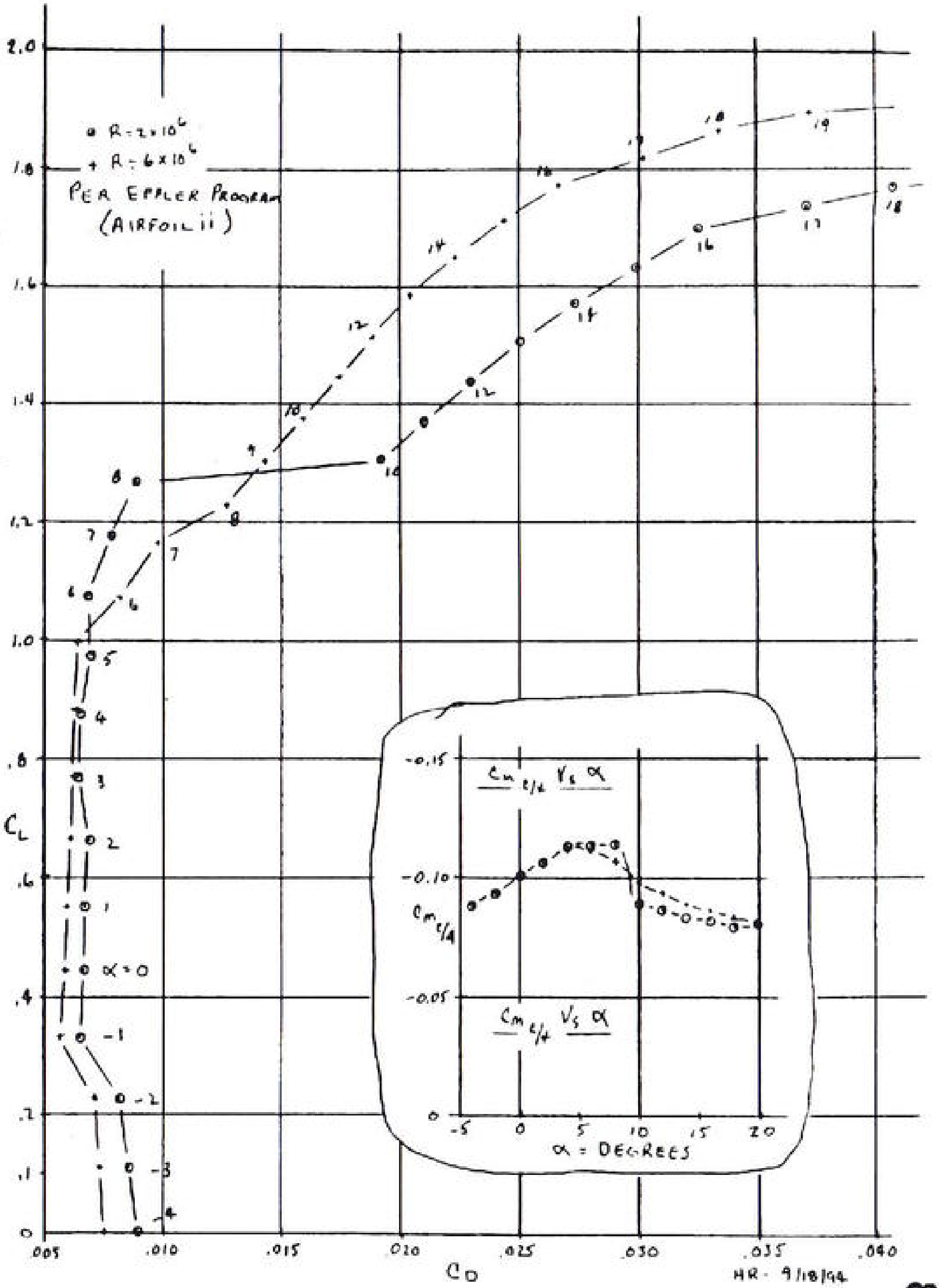


HR - 2/20/94

GA 37A 415 AIRFOIL

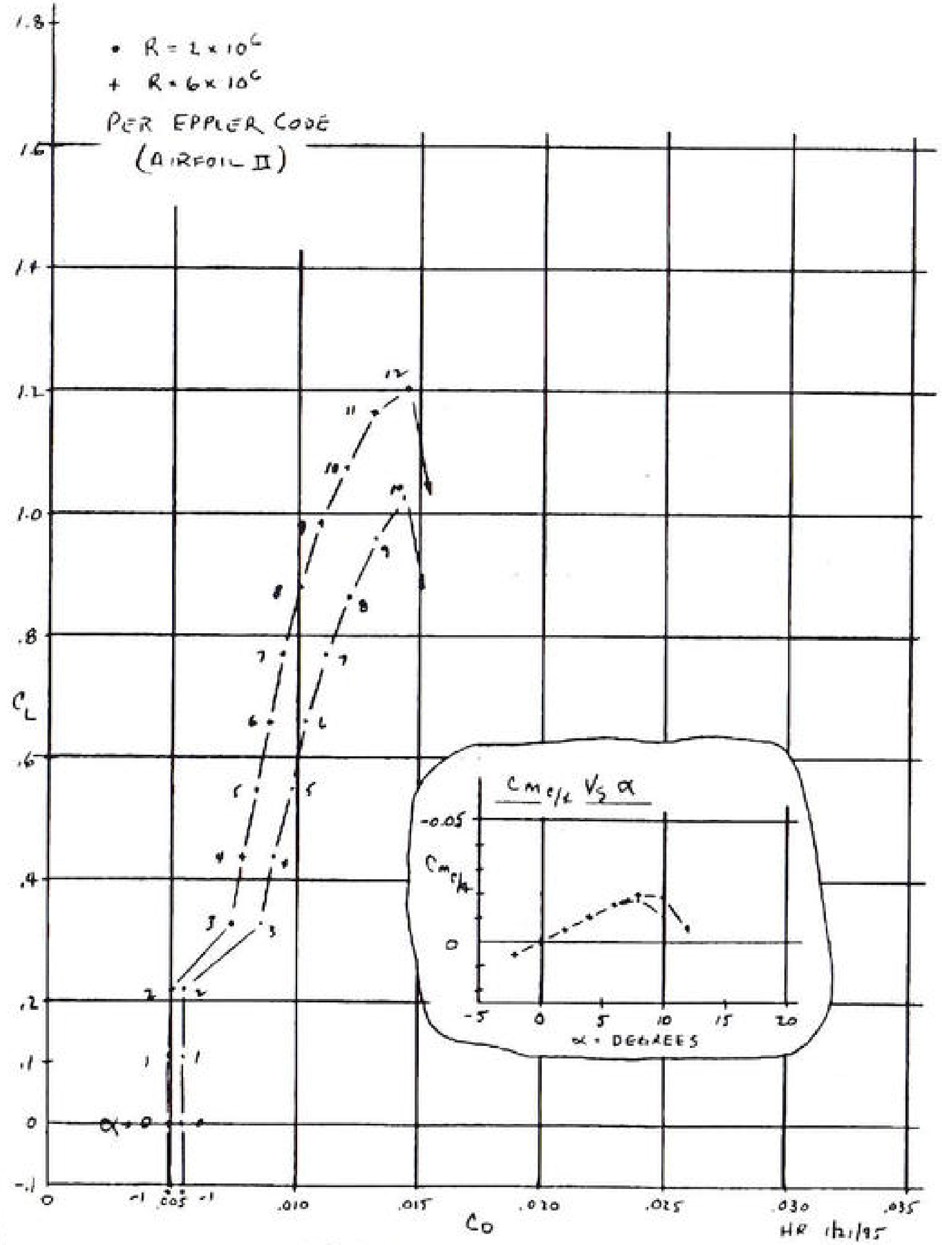


GA 37A612 AIRFOIL

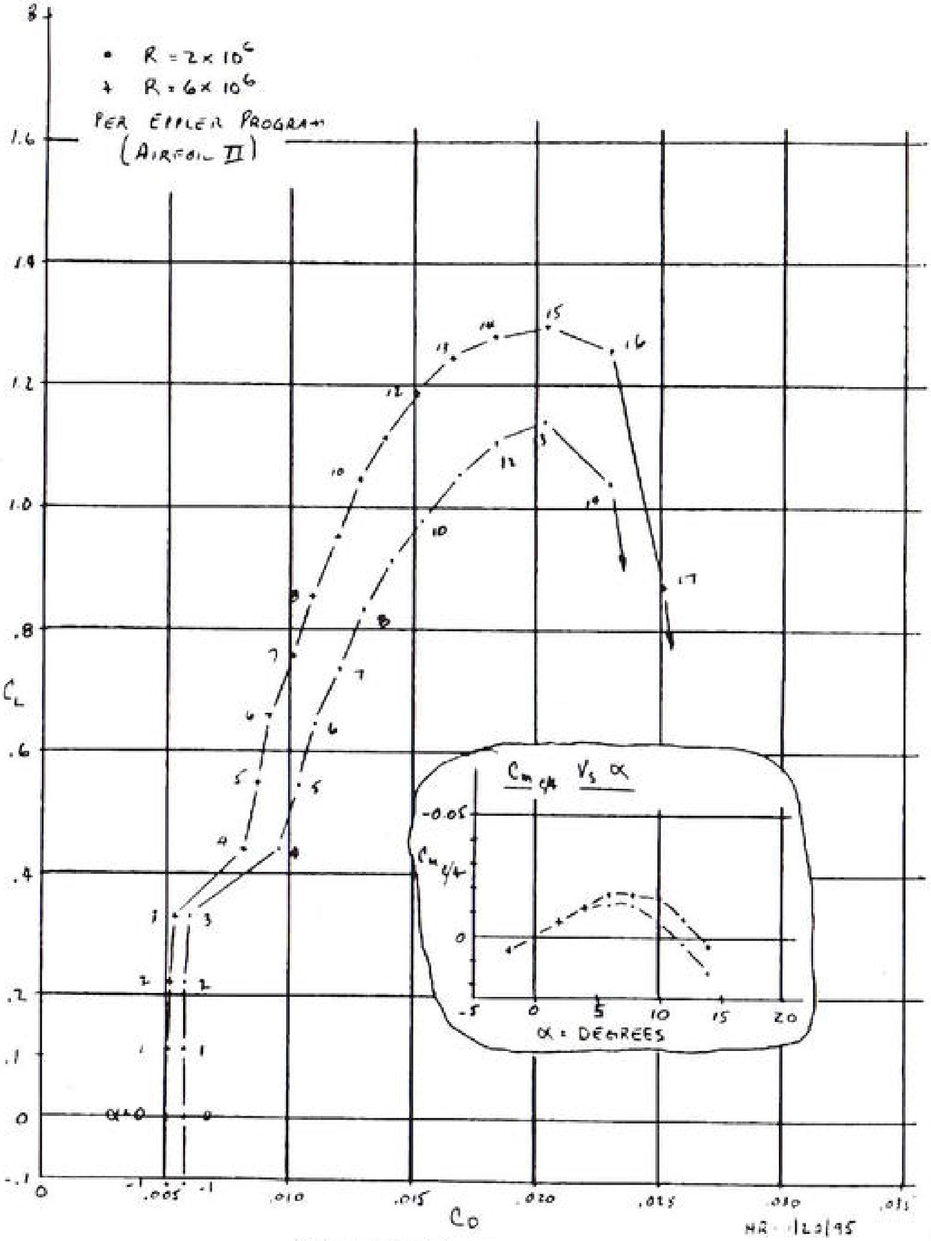


GA 37 A615 AIRFOIL

FIG IV-15

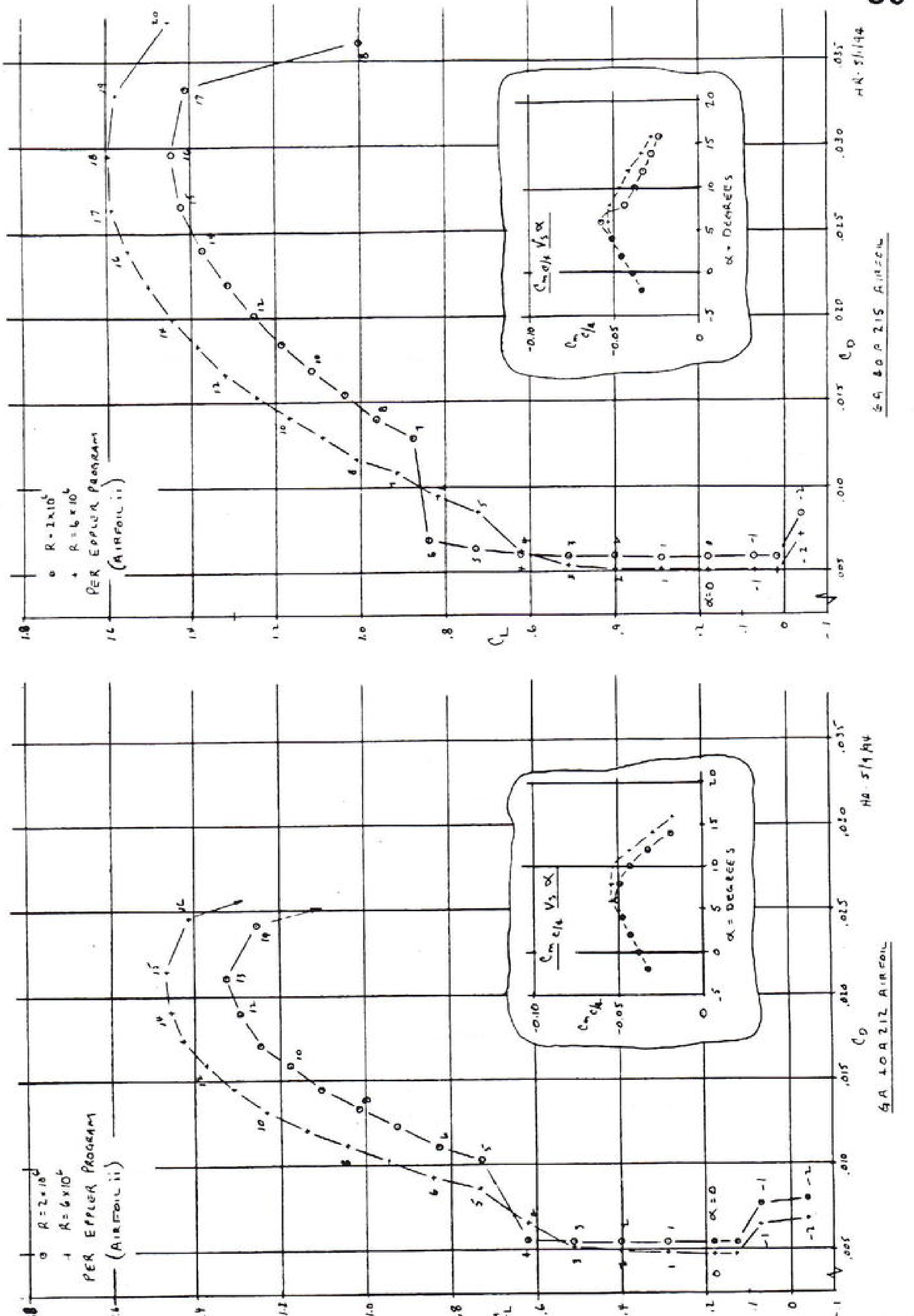


GA 40A012 SYMMETRICAL SHAPE



GA 40A015 SYMMETRICAL SHAPE

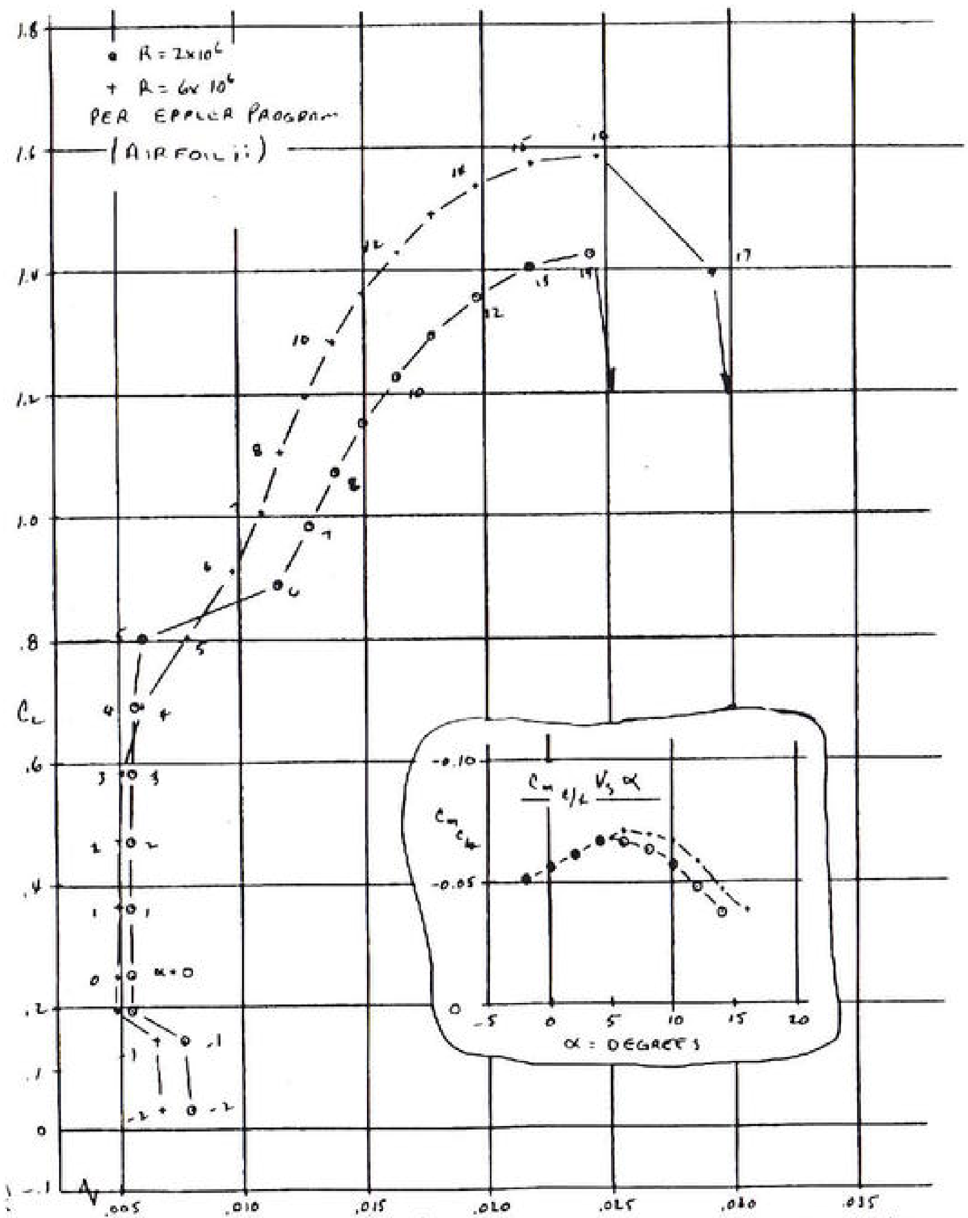
Fig. III-16



GA 40P215 AIRFOIL

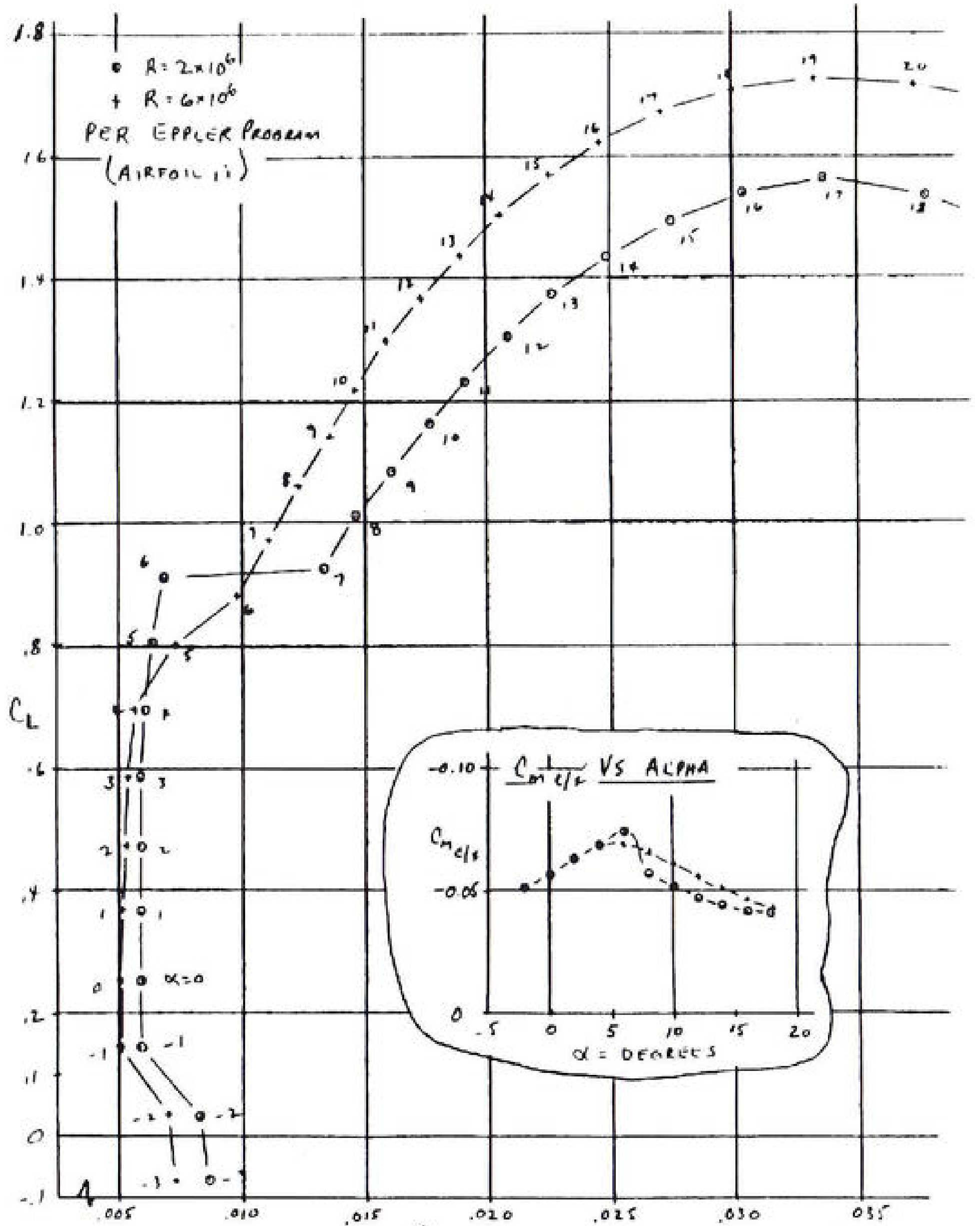
GA 40A212 AIRFOIL

FIG IV-17



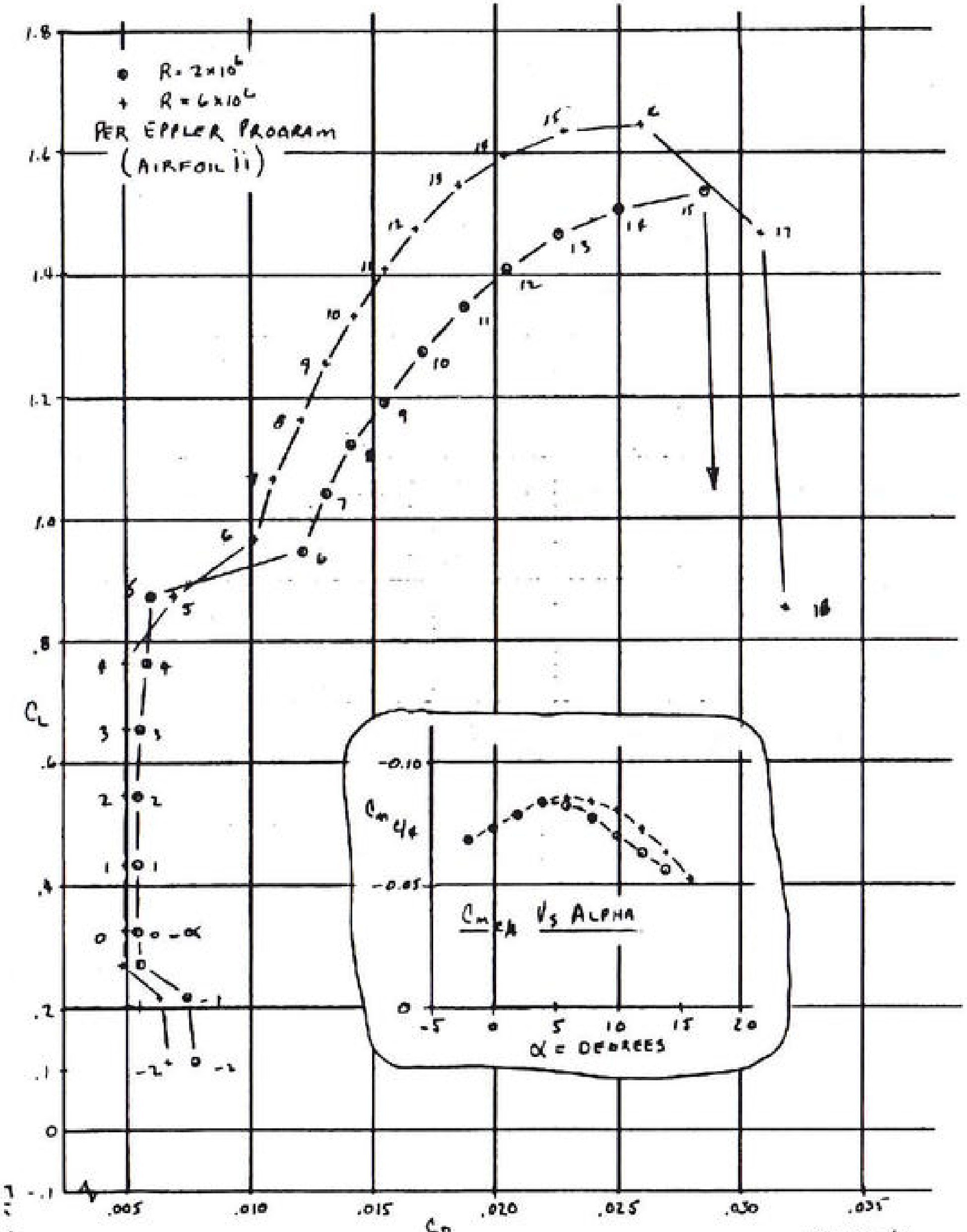
HR-5/9/94

GA 40A312 AIRFOIL



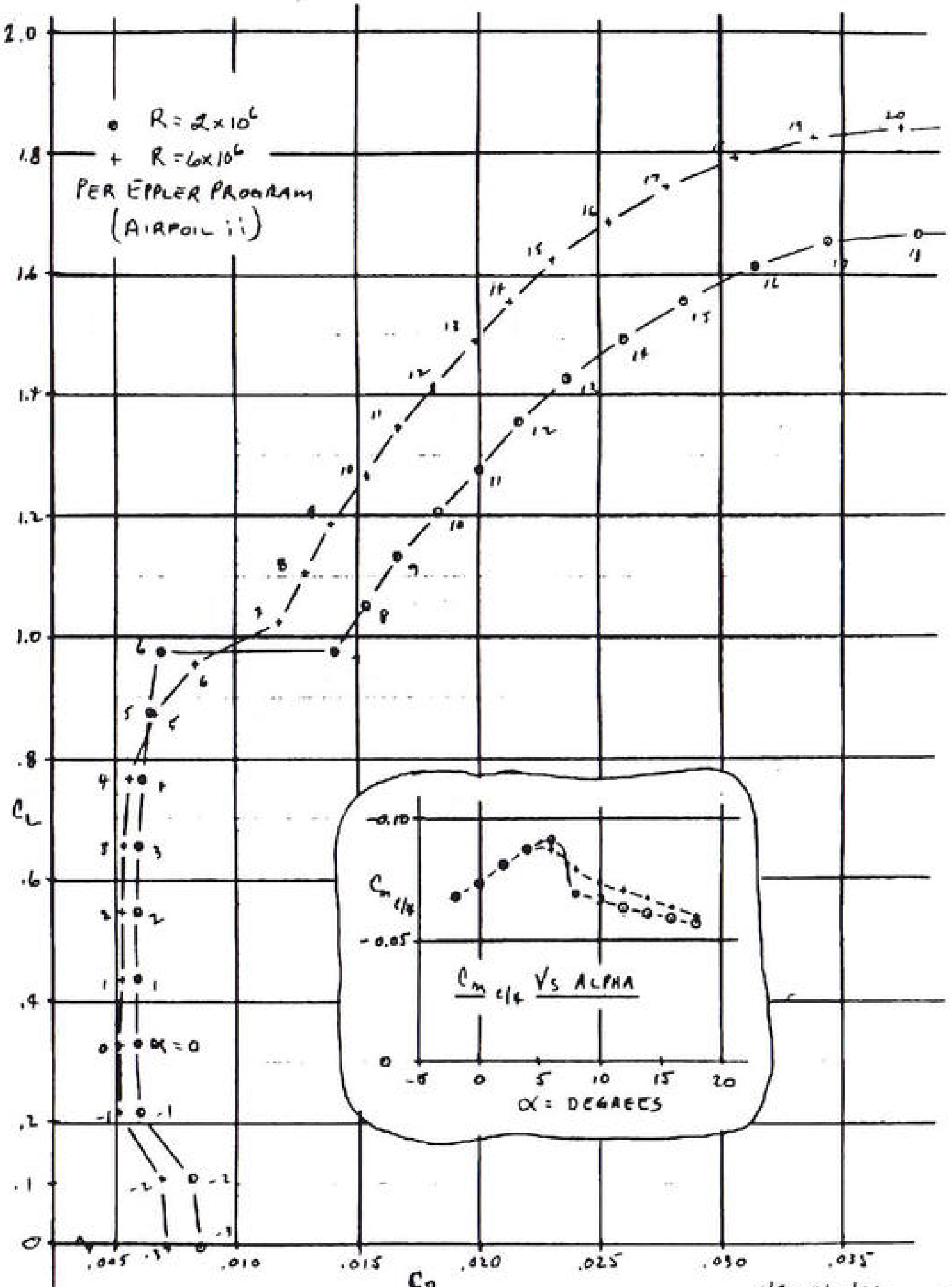
HR-4/21/94

GA 40A315 AIRFOIL



HR - 4/21/94

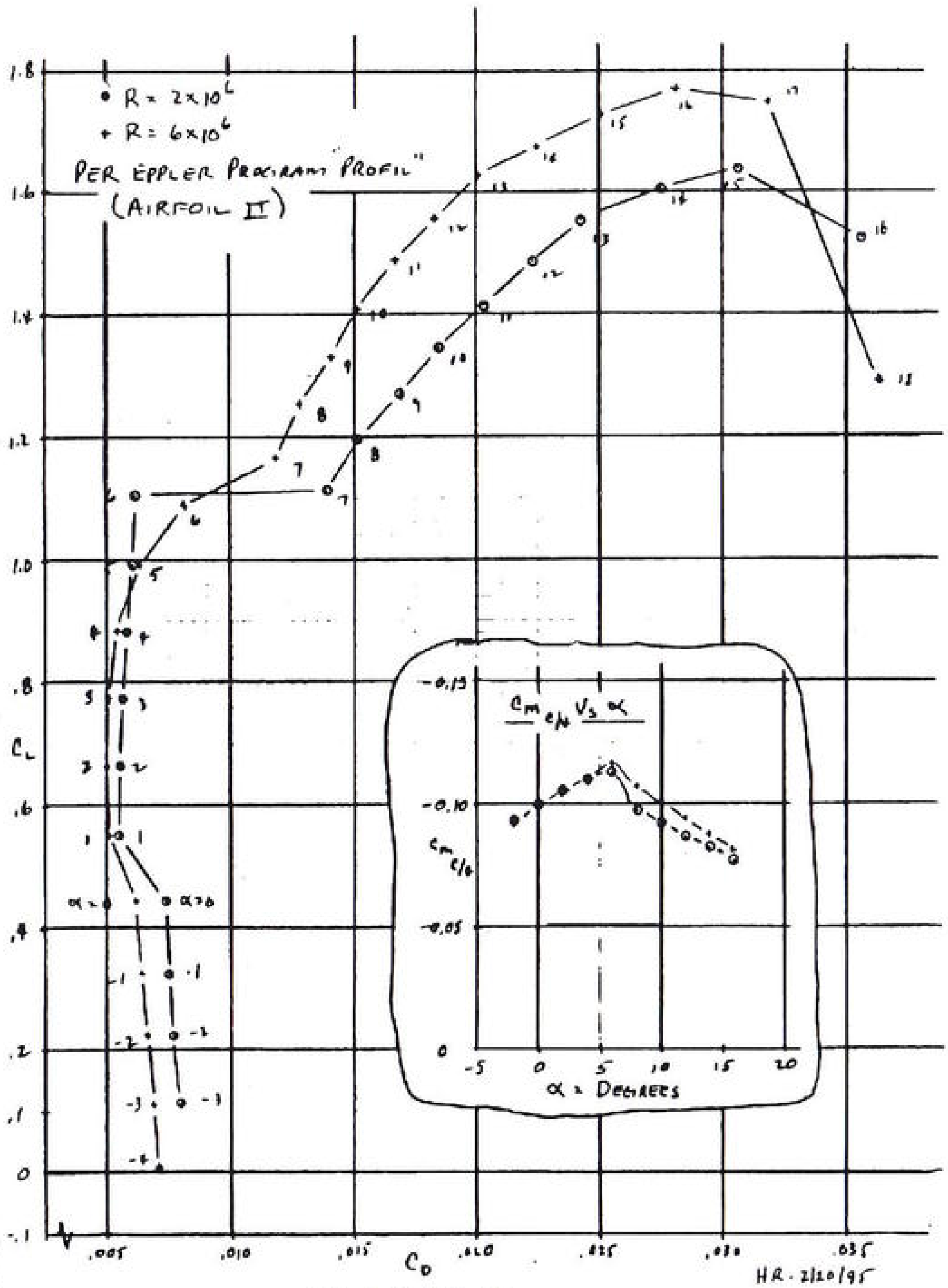
GA 40A412 AIRFOIL



HR - 4/21/94

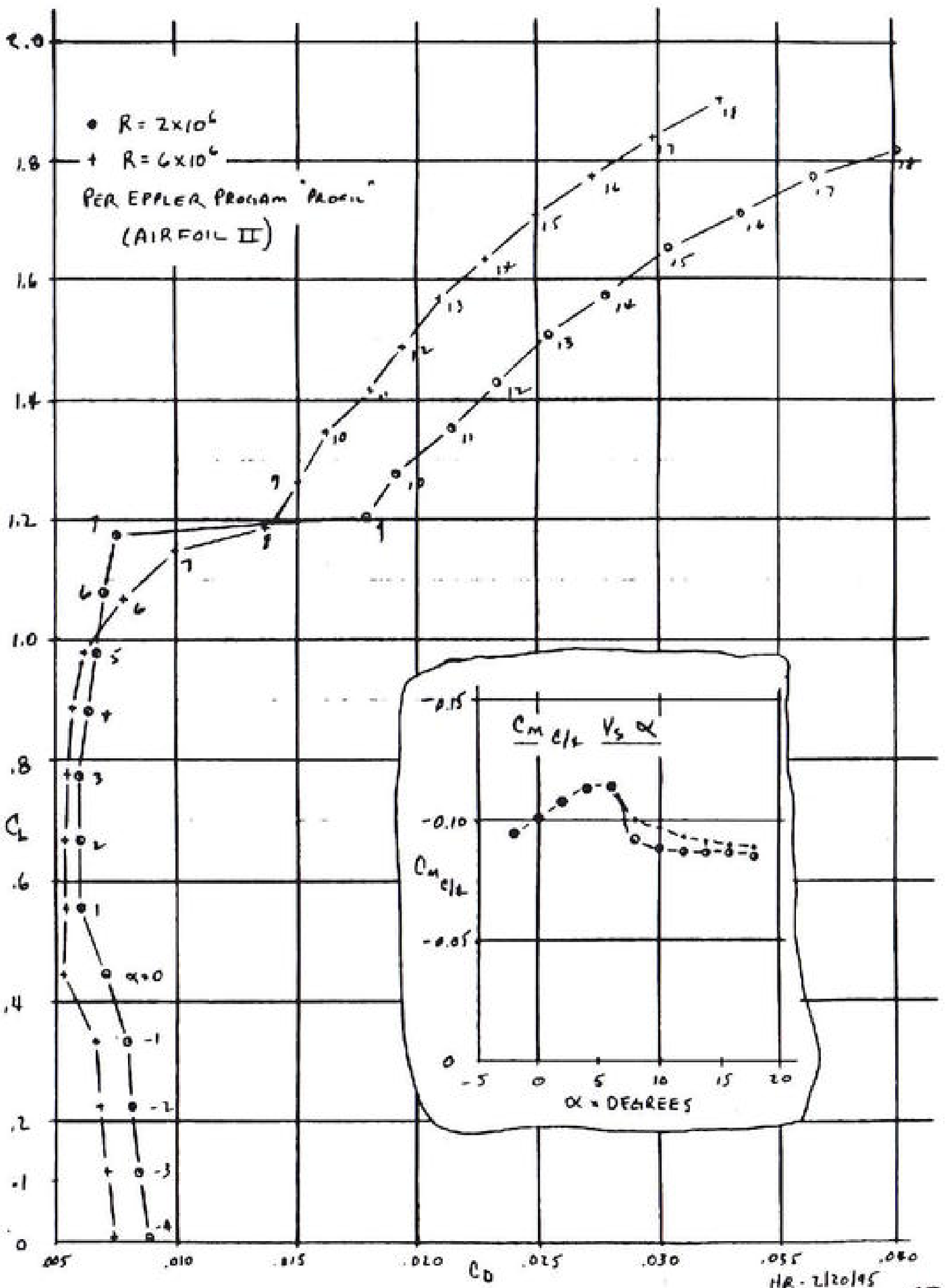
GA 40A415 AIRFOIL

Fi. III-14



HR-2/20/95

GA 40A612 AIRFOIL



HR-2/20/95

GA 40A615 AIRFOIL

FIG IV-20

Understanding Airfoils

Or, "Everything you always wanted to know about airfoils but were afraid to ask", for fear of feeling stupid, perhaps. Don't feel bad- there are some confusing aspects about airfoil performance, and a surprising number of stupid airfoil design mistakes have been made in the past by people who should have known better. We will point out a few of these, but the primary purpose of this article is to "get back to basics" so that you can better understand the main causes and effects of subsonic airfoil performance.

Symmetrical Airfoil Performance. We'll start with a simple case of two symmetrical (uncambered) airfoils, 15% thick, shown on figure 1. The first of these "basic thickness forms", NACA 0015, is considered a "turbulent flow" shape, while the second, NACA 64A015, developed later, is a "laminar flow" type shape. We drill a hole in each of these at the quarter-chord point to receive an axle, and then mount them (one at a time) in a wind tunnel such that they are free to pivot on this axle, and then blow on them. The airfoil is then tested at various angles of attack (α) from zero to about 18 or 20 degrees, plus and minus, in one degree increments, and lift and drag are recorded at each angle of attack. Lift and drag are easy to visualize, measured in pounds, acting at the pivot point. Lift is measured perpendicular to the air stream, and drag is measured parallel to the air stream. The lift and drag values at each angle of attack are now converted to dimensionless performance coefficients, Cl and Cd, considering the wing area, air density, and air speed. We next plot Cl versus Cd to obtain the "lift curves" shown on figure 3. The Cl vs Cd curve is symmetrical about alfa = 0, where Cl = 0, since the airfoil itself is symmetrical. The zero-lift drag is called "profile" or "form" drag, and the total drag increases as angle of attack increases, due to "induced" drag, or drag due to lift. As the angle of attack increases beyond about 15 degrees, the lift begins to decrease as flow separation occurs on the top surface, beginning at the trailing edge. This area of separated flow progresses forward from the trailing edge as the angle of attack increases, causing further loss of lift, and large increases of drag, until the airfoil eventually "stalls".

We are not finished with our wind tunnel measurements, however. On some airfoils the lift and drag are not exactly centered on the quarter-chord point, that is, the "center of pressure" may be some distance forward or aft of the quarter-chord point, and the airfoil tends to rotate about the pivot point. For this reason, we must also mount a spring scale at the trailing edge of the airfoil (see figure 1) to measure the pitching moment in foot-pounds about the quarter-chord point. A nose-down pitching tendency is, by convention, a negative pitching moment. We now convert the pitching moment (spring scale reading times .75C) at each angle of attack to a dimensionless coefficient, Cm, similar to Cl and Cd, by considering the wing area, air density, air velocity, and also chord length. When we plot the pitching moment coefficient curve, $Cm_{c/4}$ vs alfa (figure 3) for these two symmetrical airfoils, we see that the Cm is zero for any angle of attack up to the stall point, that is, the center of lift (center of pressure) passes directly through the quarter-chord point. Indeed, the center of lift for any symmetrical airfoil is always at the quarter-chord point, regardless of the angle of attack. Thus, this point has special significance, which is the reason that we picked it for the mounting point for wind tunnel tests in the first place. The explanation for this phenomenon is beyond the scope of this article, which is another way of saying that I don't know why it is. Regardless, the pitching moment coefficient about the quarter-chord point for any symmetrical airfoil is always zero.

Laminar Flow. The next thing to look at is laminar flow. The air all around us in itsnatural, undisturbed state is laminar, that is, in layers. If you move your hand slowly in front of you, horizontally, the air remains in layers, and your hand passes through smoothly with minimum drag, without disturbing the layers. This is known as "laminar flow". If you move your hand rapidly enough, or if your hand is rough enough, the flow over your hand will "trip" to turbulent flow, which has more than twice as m uch drag as laminar flow. The requirement for laminar flow is that the flow over the surface should not be accelerated beyond certain limits in any direction- fore and aft, sideways, or vertically- that is, we need constant velocity flow over the surface. If the flow has tripped to turbulent, we need only to slow down to the point where the acceleration limits are not exceeded, and the flow will once again revert immediately to laminar.

We see that the 64A015 shape has a region of low drag between -4 and +4 degrees angle of attack. This low-drag region is known as the laminar bucket. Beyond ± 4 degrees, most of the laminar flow is lost, and the flow over the entire wing becomes turbulent. Laminar flow is easiest to maintain near the front of the airfoil where the pressure gradient is highest ("most favorable"), and is hardest to maintain beyond the thickest part of the wing. Accordingly, the "laminar" shapes have been designed for constant velocity flow over the wing at cruise, and the thickest part of the wing has been moved as far aft as practicable, to maximize the laminar run. The net result is that the laminar section has approximately 20% less drag at cruise than the comparable turbulent section, a significant performance advantage.

Some recent "slab shaped" laminar sections admittedly have lower drag coefficients than the NACA 6-series shapes, but their laminar buckets are correspondingly narrower. They cannot tolerate much change in angle of attack without tripping to turbulent flow. At higher Reynolds numbers (higher speeds, larger airplanes) laminar flow is more difficult to maintain anyway, so the bucket becomes even narrower with these airfoils, which makes them impractical for many applications. All in all, the NACA 6-series thickness distributions are not bad, having a comfortable bucket width and depth, permitting wing twist if desired, climb at relatively slow speed, etc, without running out of the bucket.

So much for symmetrical shapes, turbulent and laminar. Next comes camber, or wing curvature.

Cambered Airfoil Performance. If we take the two symmetrical shapes discussed above, and curve them upwards in the middle, we obtain the two cambered airfoils shown in figure 2. The camber schedule is carefully designed, and is called the "mean line". The camber profile for the NACA 2415 is a simple parabola, described in reference (A). The camber schedule for GA37A315 is a modification of another mean line from reference (A), and is especially designed for laminar flow. The maximum camber in each airfoil is about the same. Also, both airfoils are forwarded loaded, that is, the maximum camber is located forward of the airfoil mid-point, at about .40C.

The performance curves for these two airfoils is shown on figure 4. Comparing with figure 3, the first thing we notice is that camber produces a significant increase of maximum lift, with no change in cruise drag. Second, the stall becomes softer, that is, the top of the lift curve is flatter, and lift continues to be developed at higher angles of attack than with the uncambered airfoils. This is primarily a function of the initial slope of the mean line (slope at .25%C), an important parameter for good slow-speed performance. This initial mean line slope has been optimized at about 15 degrees on all "GA" airfoils, correcting an unfortunate omission in the NACA work. The third thing that we notice is that cambering lifts the whole performance curve (Cl vs Cd) vertically upwards into a more usable range of Cl, with no increase in drag. Airfoil performance is meaningless below Cl=.15, except for a few very high performance airplanes with extremely high power to weight ratios. All other airplanes operate above this point: for example, most light GA airplanes operate at cruise lift coefficients between Cl=.2 and .4. We see that our cambered airfoil performance curves, and especially the laminar bucket, are now centered vertically on the design lift coefficient (Cl=.3) instead of being centered on Cl=0. Thus the top edge of the laminar bucket is raised to nearly Cl=.8, well into the climb range. The bottom edge of the bucket is at Cl=0, remaining comfortably below the lowest operating Cl minimum of Cl=.15. All good stuff.

There is one big disadvantage caused by the camber, however. Notice that the pitching moment coefficient is no longer zero, but is Cm=-.05 for both airfoils. Increasing the camber, or moving the loading aft, both result in an increase in Cm. Thus, a pitching moment is the price we pay for cambering the airfoil, and it requires us to investigate the effects of Cm on airplane performance so that we can design our airfoils intelligently.

Trim Drag. As shown on figure 5, the center of lift (center of pressure) for cambered airfoils does not always remain at the magic quarter-chord point. It does so at the stall, but as the angle of attack decreases, the center of lift moves rearward. But since we must always locate the aircraft C.G. at or near the stall C.P,, so that the nose of the airplane will drop at the stall, the wing lift vector at cruise for cambered airfoils is aft of the weight vector, and the airplane has a nose-down pitching tendency. The fix? We put a horizontal tail surface on the airplane to react the pitching moment, holding the tail down. But as we learned earlier, negative lift in the tail surface produces induced drag, which adds to the wing drag. Remember also that for every pound of negative lift in the tail, an additional pound of lift must be generated by the wing so that the summation of vertical forces remains zero. For example, if the airplane weighs 1000# and the tail down-load is 40#, the wing must generate 1040# total lift. This additional 40# of lift, above what would be required for our symmetrical airfoil, also creates induced drag. So we see that trim drag, or the drag associated with trimming out the pitching moment of a cambered airfoil, has two components- the induced drag of the tail, and the additional drag of the wing in producing an increment of lift equal to the tail down-load. For our two cambered airfoils, figure 6 shows that the trim drag at cruise is approximately 5-8 drag counts, which is about 10% of the section drag coefficient, and additive to it. Conclusion? Unless we can show at least 10% increase in maximum lift from cambering the airfoil, it isn't worth it. Happily, the increase in Clmax for our airfoils is about 30%, considerably greater than the drag increase of 10%, so we have a bargain. But this is not always the case, especially with aft-loaded airfoils, so the bigger lesson is that we should always consider the effect of differences in Cm on airplane performance when comparing airfoils. We also see that it behooves us, when designing subsonic (light GA) airfoils, to keep the Cm as low as possible, other things being equal. Using smooth, forward-loaded mean lines, and using only as much camber as necessary, does this. This is a major criticism of the NACA 6-series airfoils, which are mid-loaded, and the infamous NA$A GAW (LS-1) airfoils, which are aft loaded, an inexcusable mistake. Don't use them. For comparison, Cm=-.05 is low, Cm=-.10 is high, and the Cm of the NA$A GAW airfoils (more than -.15) is outrageous.

There are other things that can be done to minimize trim drag, such as using a long tail length, at least three wing chord lengths long. A high aspect ratio horizontal tail surface also helps. Another effective technique is to have a weight shift to the rear at cruise, following the C.P. shift to the rear. Retracting the landing gear to the rear does this, as on the Lancair IV and the Cessna 210.

Summary. There are three coefficients of airfoil performance- Cl, Cd, and Cm- equally important. The effect of airfoil Cm on airplane performance must be quantified and not neglected in designing airfoils. Only forward-loaded mean lines, such as the NACA (0.5) mean line of reference (A), should be used for common subsonic GA airfoil designs, to minimize Cm. It is not necessary to reduce the airfoil Cm completely to zero, which was done with the NACA 5-digit (230xx) airfoils,at the unfortunate cost of degrading safety, but only to keep the Cm as low as practicable, other things being equal.

Disclaimer. This paper is admittedly a simplified explanation of airfoil performance and design. For example, wind tunnel force measurements are now commonly taken with pressure rakes rather than old-fashioned spring scales. Also, for satisfactory horizontal aircraft stability, it is not necessary that the tail lift coefficient be negative, merely that it be less than the wing lift coefficient, provided that the many other factors affecting horizontal stability contribute in a favorable (stabilizing) direction. However, at slow flight, which is normally the most critical condition for longitudinal stability, the center of pressure of any airfoil approaches the C/4 point, and the conclusions presented herein regarding "trim drag" or drag associated with trimming out the wing Cm, are not invalid.

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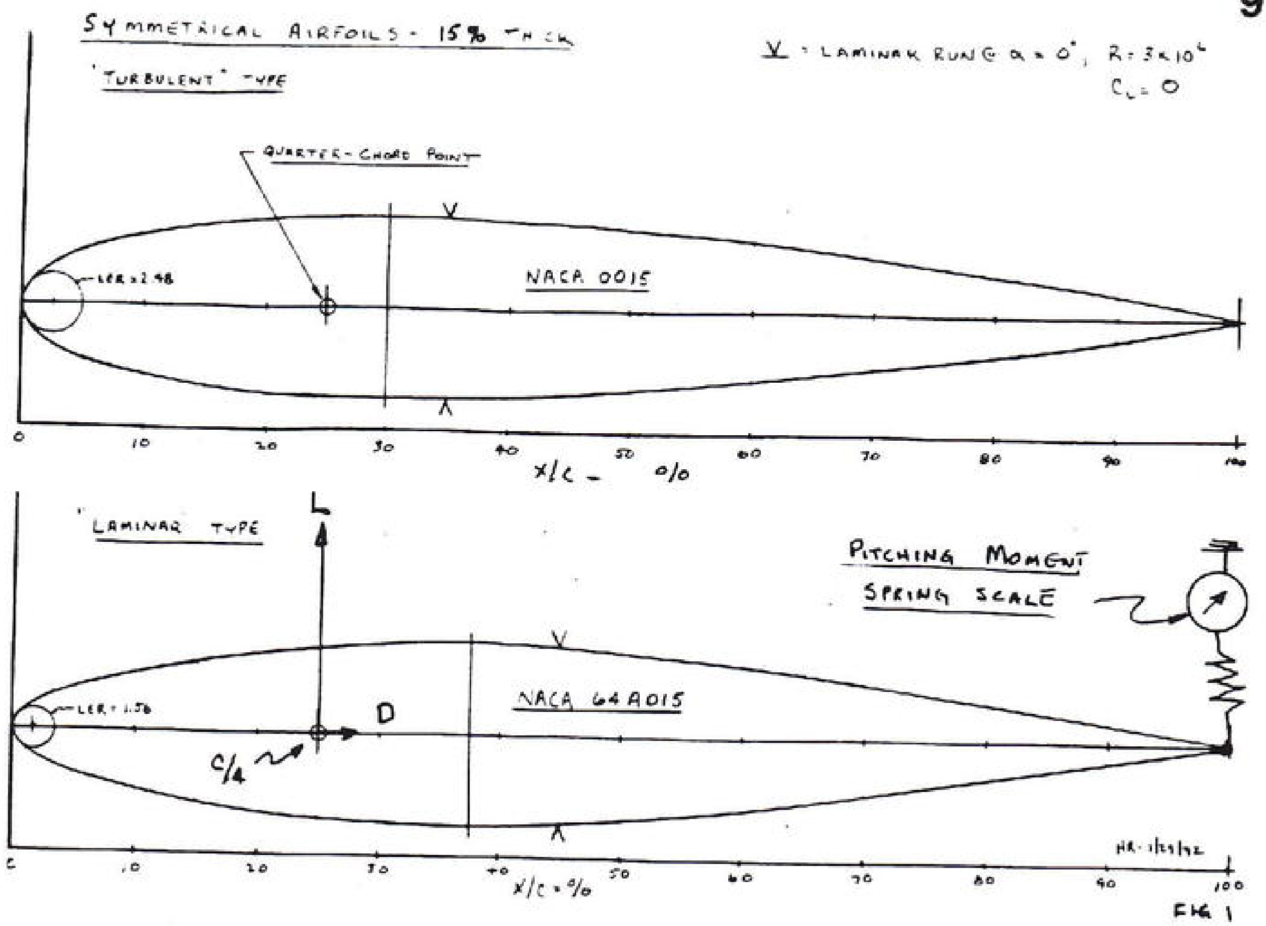

SYMMETRICAL AIRFOILS - 15% THICK
'TURBULENT' TYPE
V - LAMINAR RUN @ α = 0°, R = 3×10⁶
$C_L = 0$
QUARTER-CHORD POINT
LER = 2.48
NACA 0015
V
Λ
0
10
20
30
40
50
60
70
80
90
100
X/C - %
'LAMINAR' TYPE
L
D
C/4
LER = 1.56
NACA 64A015
PITCHING MOMENT
SPRING SCALE
HR - 1/29/92
X/C = %

FIG 1

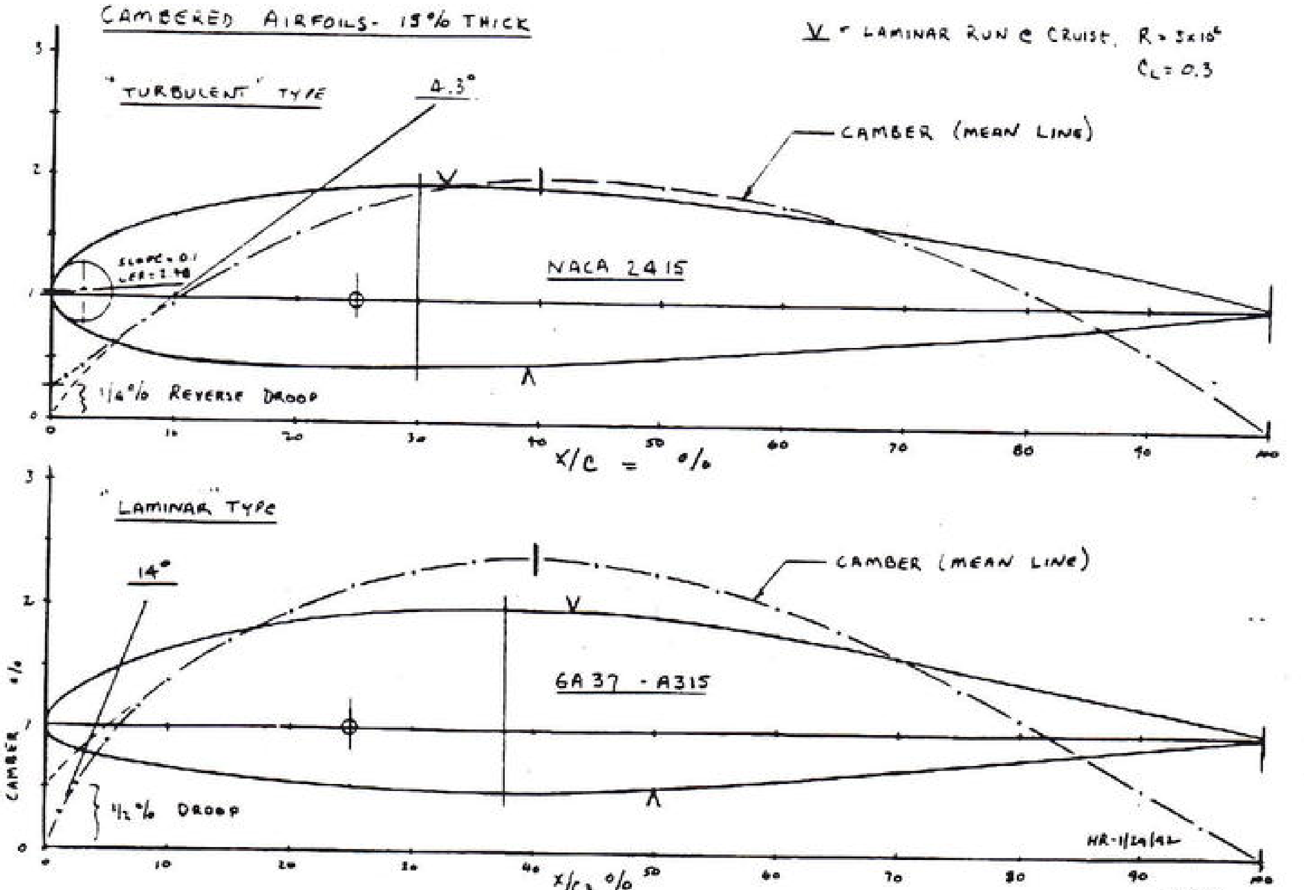

CAMBERED AIRFOILS - 15% THICK
"TURBULENT" TYPE
V - LAMINAR RUN @ CRUISE, R = 3×10⁶
$C_L = 0.3$
4.3°
CAMBER (MEAN LINE)
SLOPE = 0.1
LER = 2.48
NACA 2415
1/6% REVERSE DROOP
3
2
1
0
10
20
30
40
50
60
70
80
90
100
X/C = %
"LAMINAR" TYPE
14°
CAMBER (MEAN LINE)
6A37 - A315
1/2% DROOP
CAMBER %
HR - 1/29/92
X/C = %

FIG 2

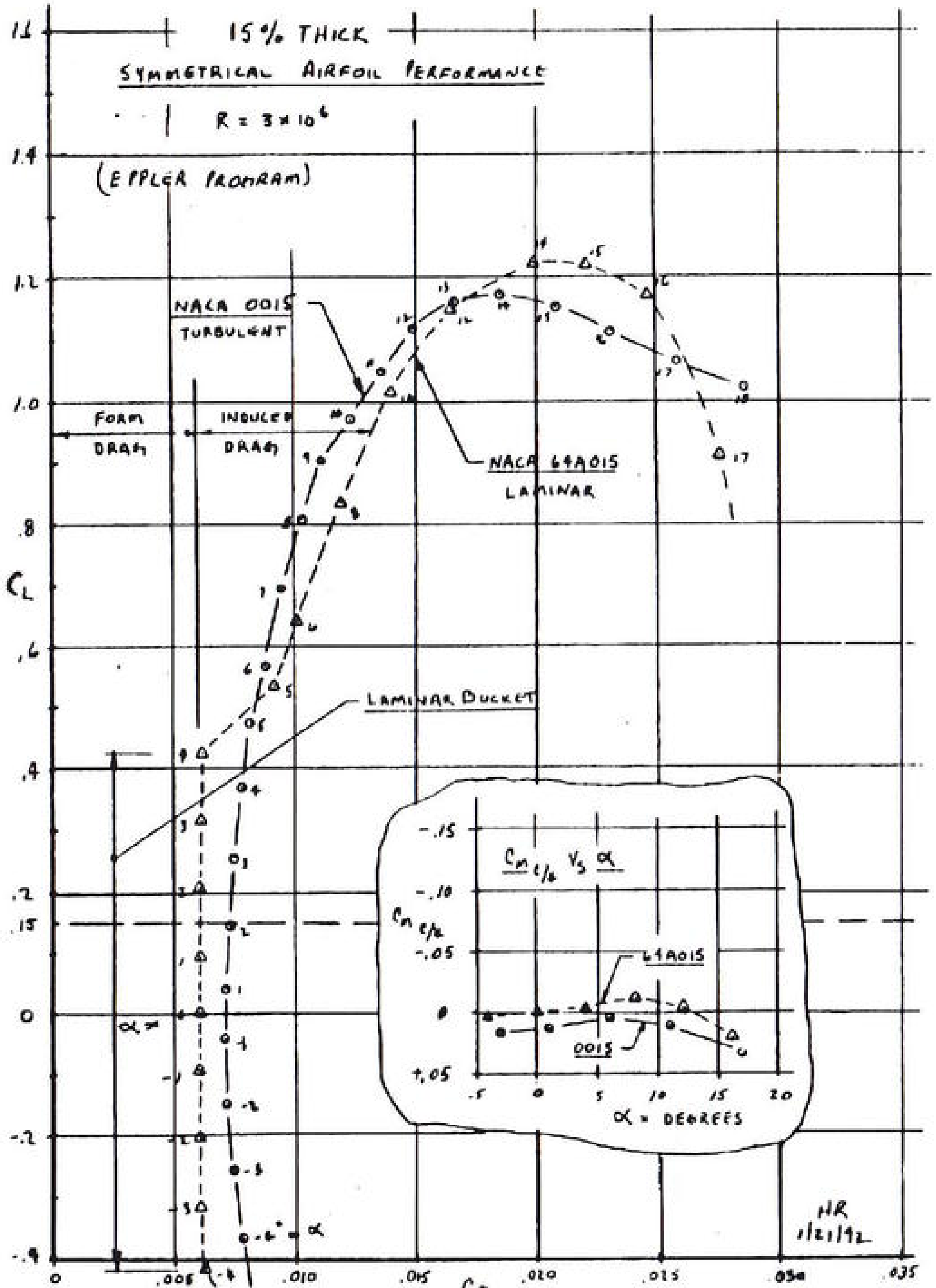

15% THICK
SYMMETRICAL AIRFOIL PERFORMANCE
R = 3×10^6
(EPPLER PROGRAM)
NACA 0015
TURBULENT
NACA 64A015
LAMINAR
FORM DRAG
INDUCED DRAG
LAMINAR BUCKET
α =
0-4° = α
C_L
C_D
1.6
1.4
1.2
1.0
.8
.6
.4
.2
.15
0
-.2
-.4
0
.005
.010
.015
.020
.025
.030
.035
$C_{m_{c/4}}$ Vs α
$C_{m_{c/4}}$
-.15
-.10
-.05
0
+.05
-5
0
5
10
15
20
α = DEGREES
64A015
0015
HR
1/21/92

FIG 3

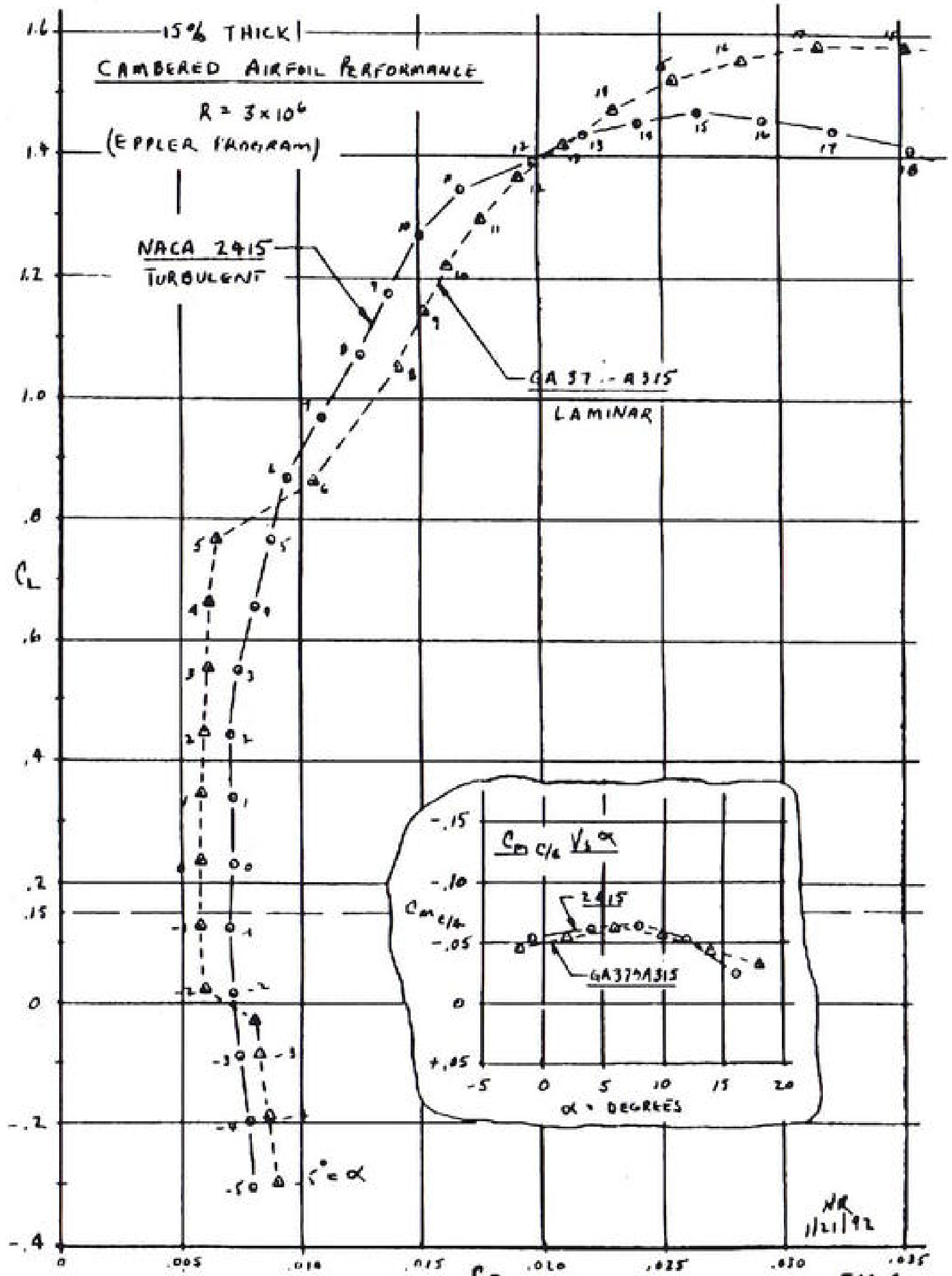

15% THICK
CAMBERED AIRFOIL PERFORMANCE
R = 3×10^6
(EPPLER PROGRAM)
NACA 2415
TURBULENT
GA 37-A315
LAMINAR
0° = α
C_L
C_D
1.6
1.4
1.2
1.0
.8
.6
.4
.2
.15
0
-.2
-.4
0
.005
.010
.015
.020
.025
.030
.035
$C_{m_{c/4}}$ Vs α
$C_{m_{c/4}}$
-.15
-.10
-.05
0
+.05
-5
0
5
10
15
20
α = DEGREES
2415
GA37-A315
HR
1/21/92

FIG 4

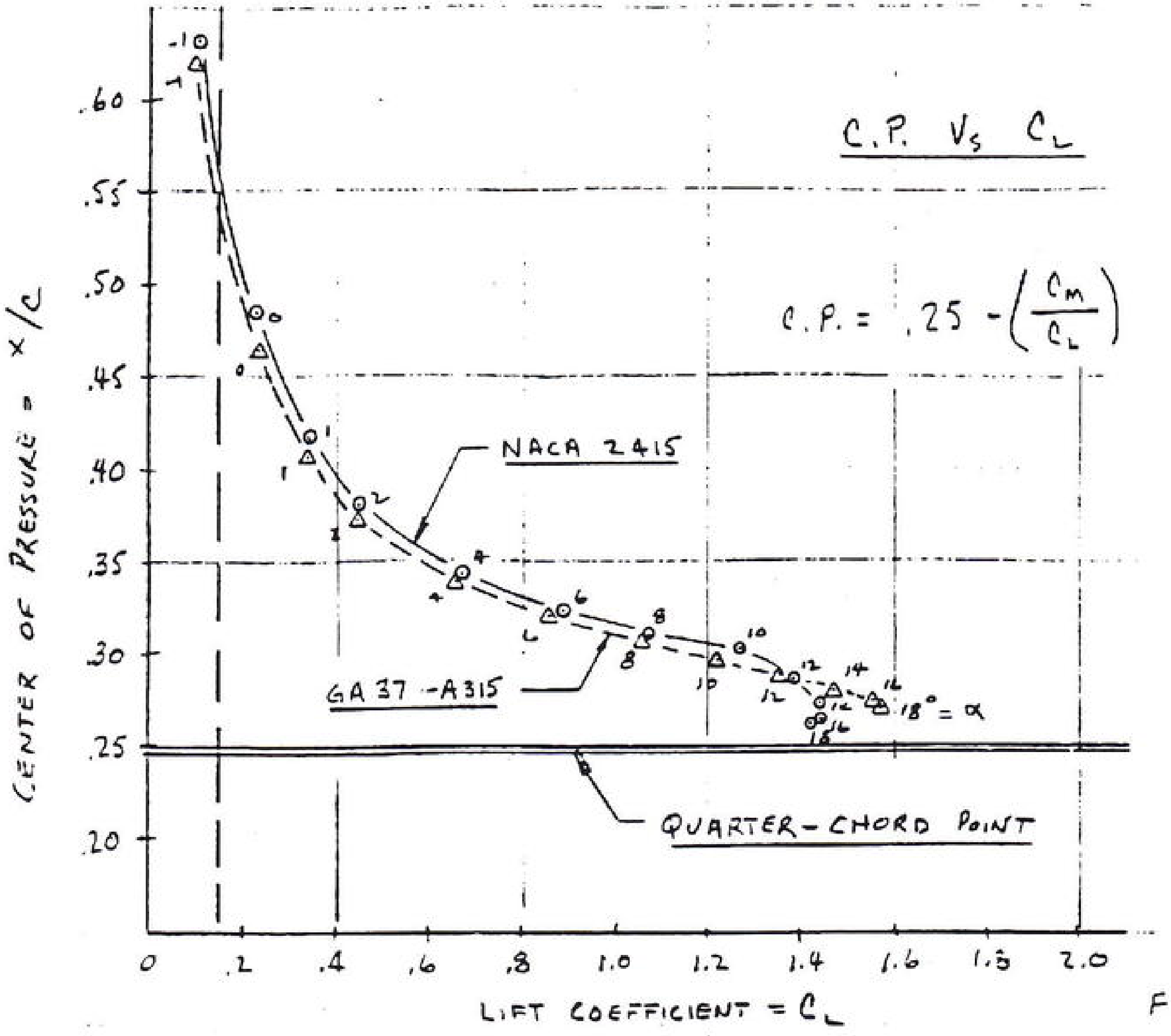
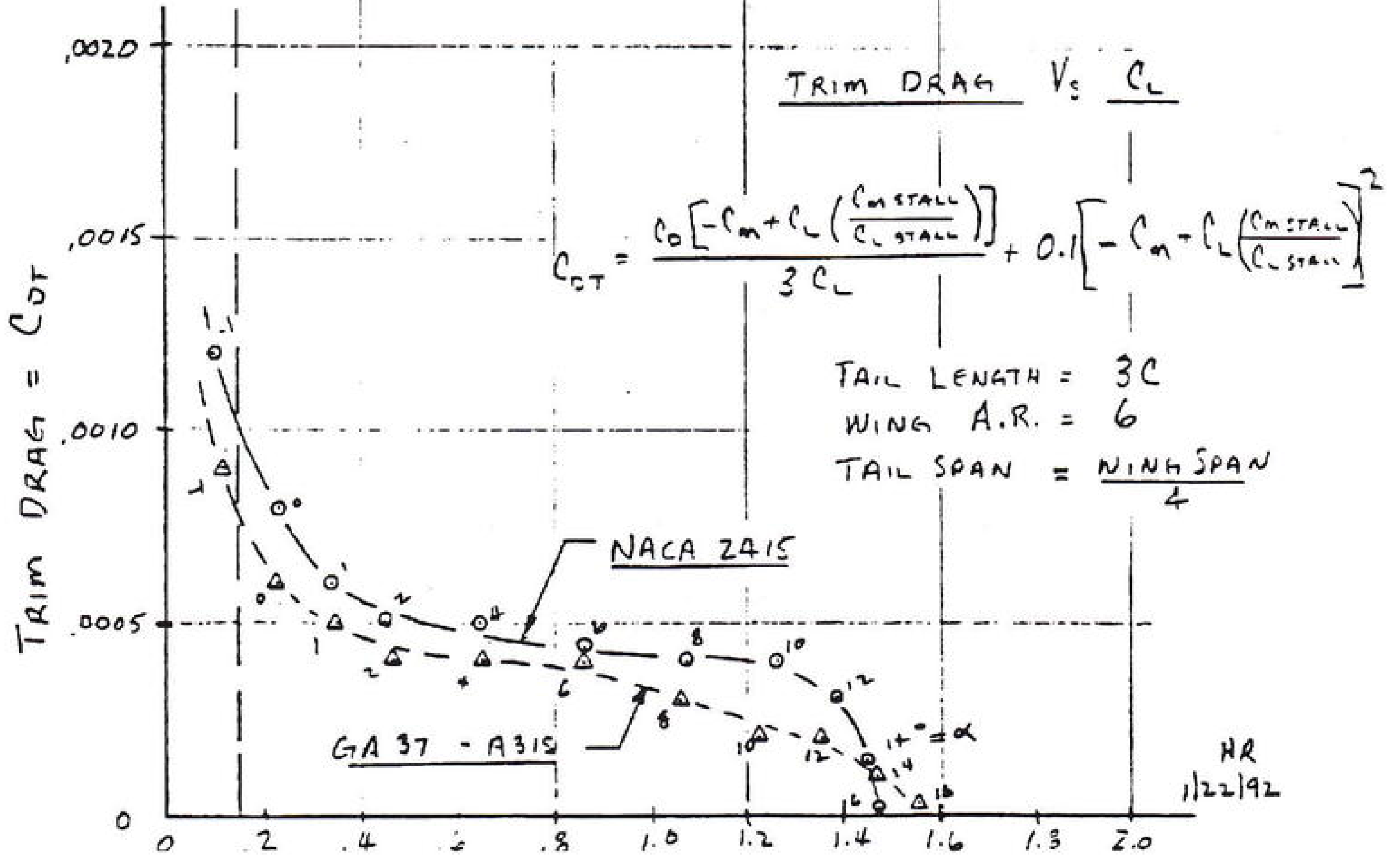


FIG 5



HR
1/22/92

ADDENDUM NUMBER 2

Design Notes for Tapered Wings

When designing a tapered wing airplane, the designer is faced with certain choices, including:

1. What airfoil sections should be used for the root and tip sections, especially concerning per cent thickness?
2. How much twist (washout) should be put in the wing to protect the wingtips from premature stall, and to preserve aileron control throughout the stall?
3. What taper ratio and what aspect ratio should be used?

Historically, designers have been guided (at least partially) by a series of tapered wing tests conducted on turbulent sections at NACA Langley in the mid-1930's. Figure 1 shows a tabulation of the specimens used during these tests, and the tabulated test results. Typical performance curves are also shown on figure 1. Note the sketch of the initial stall distributions predicted by NACA for "rectangular, tapered, and swept-back wings". This sketch is especially bothersome, since it illustrates the NACA conclusion that "rectangular wings stall first at the wing roots, and tapered wings stall first at the wing tips"- hence, supposedly, rather drastic measures are needed, such as use of considerable washout, or use of sections with more camber at the tips, and then even more washout, to combat these "bad tip stall characteristics" of tapered wings. The bothersome aspect is that the planform differences between the straight and tapered wings are not so great as to account logically for the marked differences in the location of initial stall reported by NACA. Nevertheless, generations of aeronautical students have been conditioned to accept these results as gospel, and to design accordingly. The commonly accepted washout recommendation is about 3 degrees, in spite of the fact that twist reduces the overall efficiency of the wing.

Let's look a little closer at the test specimens used in the NACA tests, however. Incredulously, of the 22 specimens tested, all but two have a % thickness taper from a root thickness of 15%, 16%, 18%, or 20% to a tip thickness of only 9%, in addition to the planform taper. The next to last specimen uses a 23013/43010 combination, that is more camber at the (thinner) tip- all the rest have the same camber, root and tip. Only the last specimen has the same per cent thickness at the root and tip, 12%, but it uses an odd (elliptical) planform. Evidently, NACA (erroneously) assumed that the preferred way to build a tapered wing is to taper both the planform and the per cent thickness, and unfortunately the test results are contaminated by the effect of these two variables combined, casting considerable doubt on the conclusions. It is curious (and unexplained) that NACA never did test the two variables separately, except for sample #22, which has an odd planform that further contaminates the data. In fact, if they had tested a 2415/2415 sample for example, with 2/1 planform taper ratio, they would have found that the initial stall distribution is the same as for a straight rectangular wing, that is, benign. Further, if they had tested a straight planform wing with a per cent thickness change from root to tip, they would have seen the location of the initial stall shift to the wing tips. Thus, contrary to NACA's conclusion, the spanwise location of the initial stall is primarily a function of the per cent thickness change from root to tip, and is not due to the planform taper. Proof of this is shown on figures 2 and 3. Note that the thin (9%) tip sections stall about 3 degrees earlier than the thicker (root) sections, for both the 24xx and the 230xx sections. Further, the most efficient section thickness in terms of maximum L/D is 12%, as shown by figure 7. Therefore, there is absolutely no reason ever to use a tip thickness less than 12%, and NACA's design of the tapered sections was a monumental mistake.

Unfortunately, following NACA's poor example, a whole generation of tapered wing airplanes was built with wing tips that were too thin. Figure 4, courtesy of Mr. George Copland of Duncan, OK, is a partial listing of representative planes from the 1930's, 1940's, and 1950's. Note the ill-advised use of 9% thick tip sections on many of these common airplanes. It is also interesting to note that a few of the more sucessful airplanes (P-38, P-51, DC-3, Bonanza) did not fall for the trap of using 9% thick tip sections, but used 12% tip sections, contrary to NACA's recommendation. Note also that the Piper PA-24 Commanche, designed in 1954, was a historically significant airplane, for it "broke the mold" by using only planform taper with no per cent thickness taper, with no washout in the wings. The result was a very docile airplane. Unfortunately, however, too many modern airplanes, such as the Venture, Malibu, and high-performance Beech models (see figure 6) continue to be built with thin wing tips, unnecessarily causing control problems and compromising safety. The 23015/23009 combination is especially ill-advised, due to the additional problem of sharp stall characteristics of the 230xx airfoils. Figure 3 shows the mechanism of this sharp, dangerous stall. The 230xx camber profile has a discontinuity (kink) in it at 15% chord. This causes the airflow, at higher angles of attack, to separate at that point, resulting in a separation bubble on the top surface, followed by reattachment before the normal trailing edge separation. The added thickness of this separation bubble on the top of the wing causes the wing to act as if it were thicker and more highly cambered (high lift) at high angles of attack, in spite of the relatively low geometric camber of the section. At low angles of attack, the bubble disappears, and the section reverts to its low-drag profile. For many years this airfoil was touted as "having the best of both worlds", but the disadvantage is a dangerously sharp stall. As the angle of attack increases further, the re-attached section gets shorter and shorter until the separation bubble suddenly expands to cover the entire top surface, resulting in a sudden loss of lift. This is bad enough on single engine airplanes, but on twin engine airplanes with one engine out the airplane may snap-roll suddenly and uncontrollably into the dead engine, if the speed drops too low. (King Air crash @ Wilmington 5/2/93, Fig 5 del) Recently, vortex generators have been used to improve this situation, as shown on figure 6. The VG's are installed at 10% of chord, just before the discontinuity at 15% of chord. They function by creating a thick turbulent (energized) boundary layer immediately aft of the VG's, preventing the separation bubble from forming. This forces the flow into an even more highly cambered shape (more lift) than with the bubble, without the disadvantage of the bubble (sharp stall). The price is a slightly lower cruise speed due to the small amount of drag from the VG's, but overall this is a very good quick fix, improving safety. Of course the better solution, next time around, is to use modern soft-stall airfoils not subject to the separation bubble problem. Also remember that the VG's are not a panacea for all airfoils, but rather a specific crutch for the poor features of the 230xx airfoils.

One final comment is needed regarding NACA sample #22, with the elliptical planform. This specimen has the highest Clmax (1.81) of any tested, and is accordingly often cited as proof of the "inherently superiority" of the elliptical planform over the uniformly tapered planforms. Note, however, that this specimen enjoys two distinct advantages over the other specimens- it has a unique high-lift section (4412), and in addition it has the same per cent thickness root and tip, 12%, which is, as we have seen, the optimum thickness in terms of best L/D. In view of this, we wonder if the legendary mystique of the elliptical planform is justified.

Regarding planform taper ratio, it is well to remember that a very short tip chord means a low Reynolds number at the tip, and airfoil section performance drops off at low Reynolds numbers. Thus it is best not to taper the wings too much. Common taper ratios go from about 3/2 to 2/1. Smaller tip chords mean smaller tip losses, but also lower Rn, as stated above. Of course a higher taper ratio (2/1) also has a structural advantage, since it permits a greater spar depth at the root for a given per cent root thickness, other things being equal. Alternatively, a higher taper ratio may permit use of a lower per cent root thickness (15% vs 18%) for better L/Dmax.

Regarding aspect ratio, the higher the better, limited only by structural considerations. Aspect ratios between 6 and 10 are common for powered airplanes.

In summary, for turbulent wings, the tips should not be less than 12% thick. For laminar wings, notice that there is little difference in L/D between 12% and 15% thick sections, thus 15% thick tip sections are preferred. If this is done, and if modern sections such as GA airfoils are used that have forward loading and leading edge droop, the tip stall problem is eliminated, or at least, is greatly reduced. Use of additional camber at the tips is not needed, nor is wing twist. A flapped section stalls at a lower angle of attack than the same section unflapped, and this in itself provides protection against tip stalls. For twin engine airplanes, to help the engine-out situation, the above recommendations are especially applicable, particularly with regard to the use of soft stall airfoils, so that assymetrical lift conditions are unlikely to develop.

NACA TAPERED WING TESTS

CHARACTERISTICS OF TAPERED WINGS, VARIABLE-DENSITY-TUNNEL TEST RESULTS*

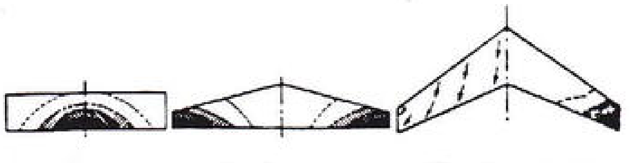
Wing†	Plan-form taper ratio	Aspect ratio	$C_{L\,max}$	α_{L0} deg	a per deg	$C_{L\,opt}$	$C_{D0\,min}$	$C_{m\,a.c.}$	a.c. fraction ahead of root $\frac{c}{4}$	e
0015-09;0-0	2:1	6	1.53	0	0.075	0.04	0.0065	0	0.014	0.89
2415-09;0-0	2:1	6	1.66	−1.7	0.074	0.14	0.0065	−0.040	−0.022	0.90
2415-09;15-0	2:1	6	1.63	−1.9	0.075	0.18	0.0065	−0.043	−0.352	0.90
2415-09;30-0	2:1	6	1.43	−1.9	0.072	0.18	0.0065	−0.042	−0.775	0.88
2415-09;30-8.5	2:1	6	1.51	0.7	0.078	0.36	0.0071	0.002	−0.786	0.92
3R₁15-09;15-8.5	2:1	6	1.59	1.2	0.078	0.36	0.0078	0.003	−0.348	0.92
3R₂15-09;15-0	2:1	6	1.50	−0.7	0.078	0.16	0.0086	0.004	−0.351	0.88
0015-09;15-2.45	2:1	6	1.48	1.0	0.078	0.06	0.0089	0.007	−0.346	0.90
0015-09;15-2.45	4:1	6	1.32	0.7	0.078	0.10	0.0070	0.005	−0.334	0.90
2218-09;0-0	2:1	6	1.60	−1.8	0.071	0.15	0.0074	−0.029	0.028	0.91
M6(18)-(09);0-0	2:1	6	1.49	−1.1	0.070	0.18	0.0071	−0.006	0.017	0.90
CYM-18-09;0-0	2:1	6	1.67	−5.2	0.071	0.22	0.0076	−0.071	0.030	0.91
23015-09;0-0	2:1	6	1.71	−1.2	0.074	0.17	0.0087	−0.007	0.014	0.91
23018-09;0-0	2:1	6	1.66	−1.3	0.073	0.11	0.0071	−0.007	0.020	0.90
23018-09;0-0	3:1	10	1.51	−1.2	0.083	0.04	0.0070	−0.011	0.013	0.80
23018-09;0-0	5:1	10	1.50	−1.2	0.083	0.03	0.0067	−0.009	0.011	0.78
23018-09;0-0	5:1	10	1.49	−1.2	0.083	0.05	0.0071	−0.011	0.013	0.81
23018-09;0-0	5:1	12	1.46	−1.2	0.086	0.02	0.0066	−0.014	0.018	0.76
23020-09;0-0	5:1	12	1.42	−1.2	0.084	0	0.0069	−0.007	0.010	0.74
0018-09;0-0	2:1	6	1.48		0.074	0	0.0069	0	0.030	0.90
23012-43010;0-2	1.6:1	6	1.67	−0.7	0.074		0.0064	−0.009		
4412-4412;0-0	Ellip.	6	1.61	−4.0	0.074	0.15	0.0071	−0.100	0.018	0.92

* Reported in NACA Tech. Rept. 627, corrected as directed in NACA Tech. Rept. 669.

† Numerical designations have the following significance: in the example 2415-09; 30-8.5, 2415 Root airfoil − 2409 tip airfoil, 30 deg sweepback on $\frac{c}{4}$ line − 8.5 deg dihedral on $\frac{c}{4}$ line.

CONCLUSIONS (?) NOT VALID

Typical stall distributions of rectangular, tapered, and swept wings are shown in Fig. 11:10, the initial stall occurring at the point of heaviest shading.



a. Rectangular b. Tapered c. Tapered and sweptback

Fig. 11:10. Initial stall distributions for rectangular, tapered, and sweptback wings.

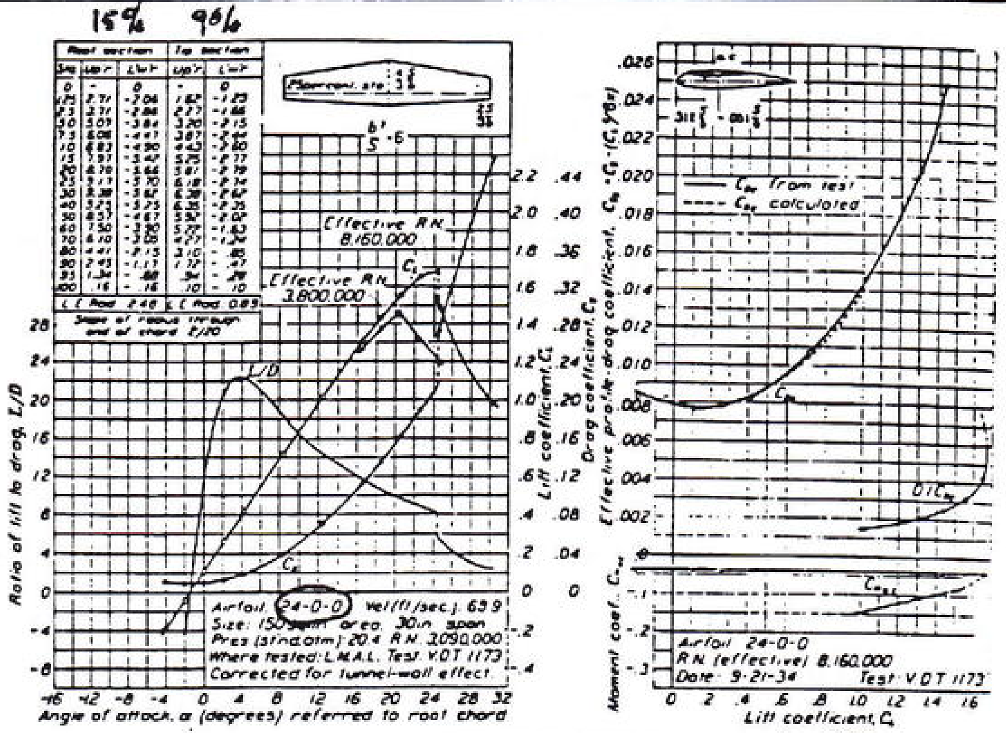
(1)Schrenk, O., "A Simple Approximation Method for Obtaining the Spanwise Lift Distribution," NACA TN 948, 1940.

(2)Peery, David, J. "Aircraft Structures." McGraw-Hill, 1950.

REF- K.D. WOOD, "AIRCRAFT DESIGN"

TYPICAL TEST RESULTS

2415 ROOT, 2409 TIP, 2/1 TAPER, PLANFORM AR=6
NACA TR 572
0° SWEEP
0° DIHEDRAL



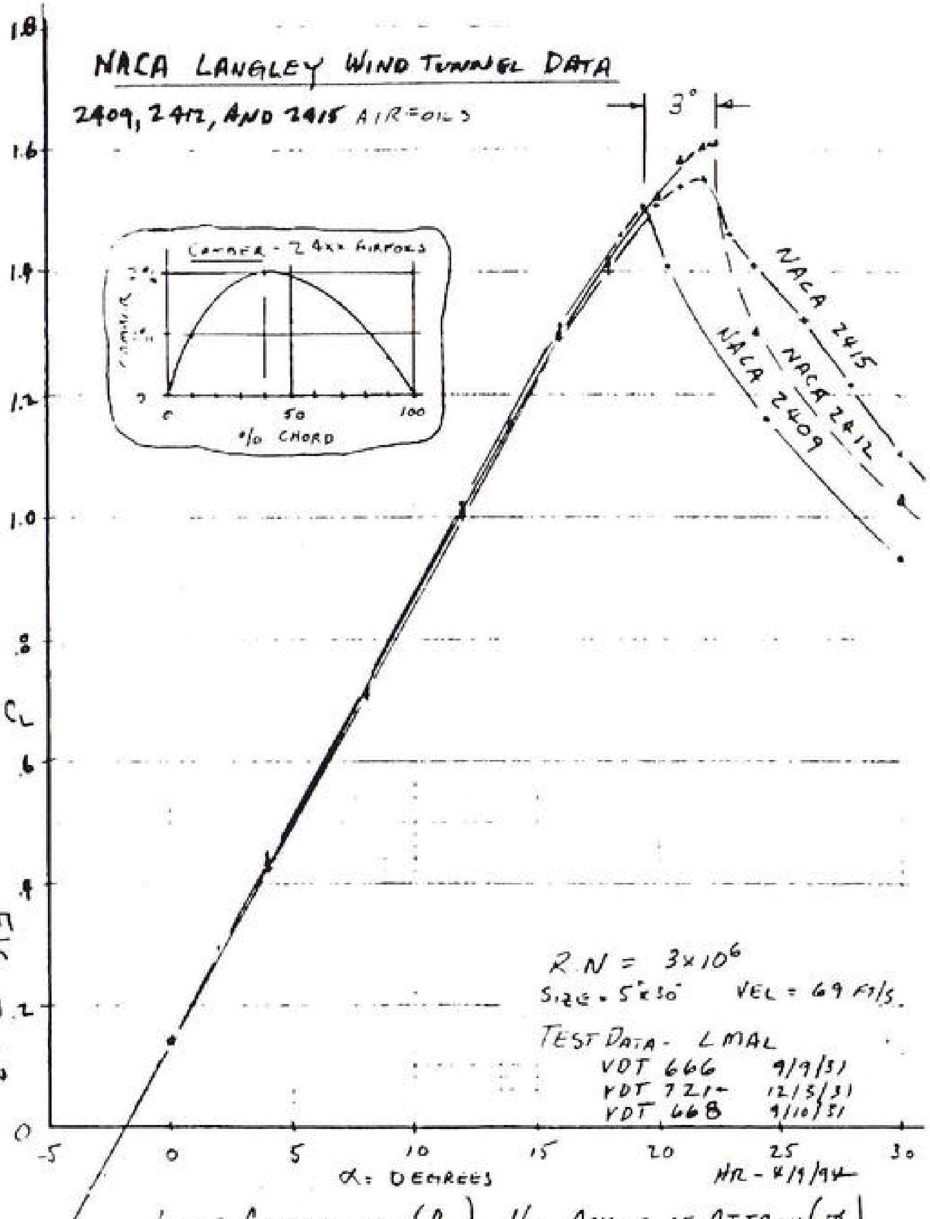
Tapered N. A. C. A. 24-0-0 airfoil.

$C_{m_{a.c.}}$ = −.0345

FIGURE 1

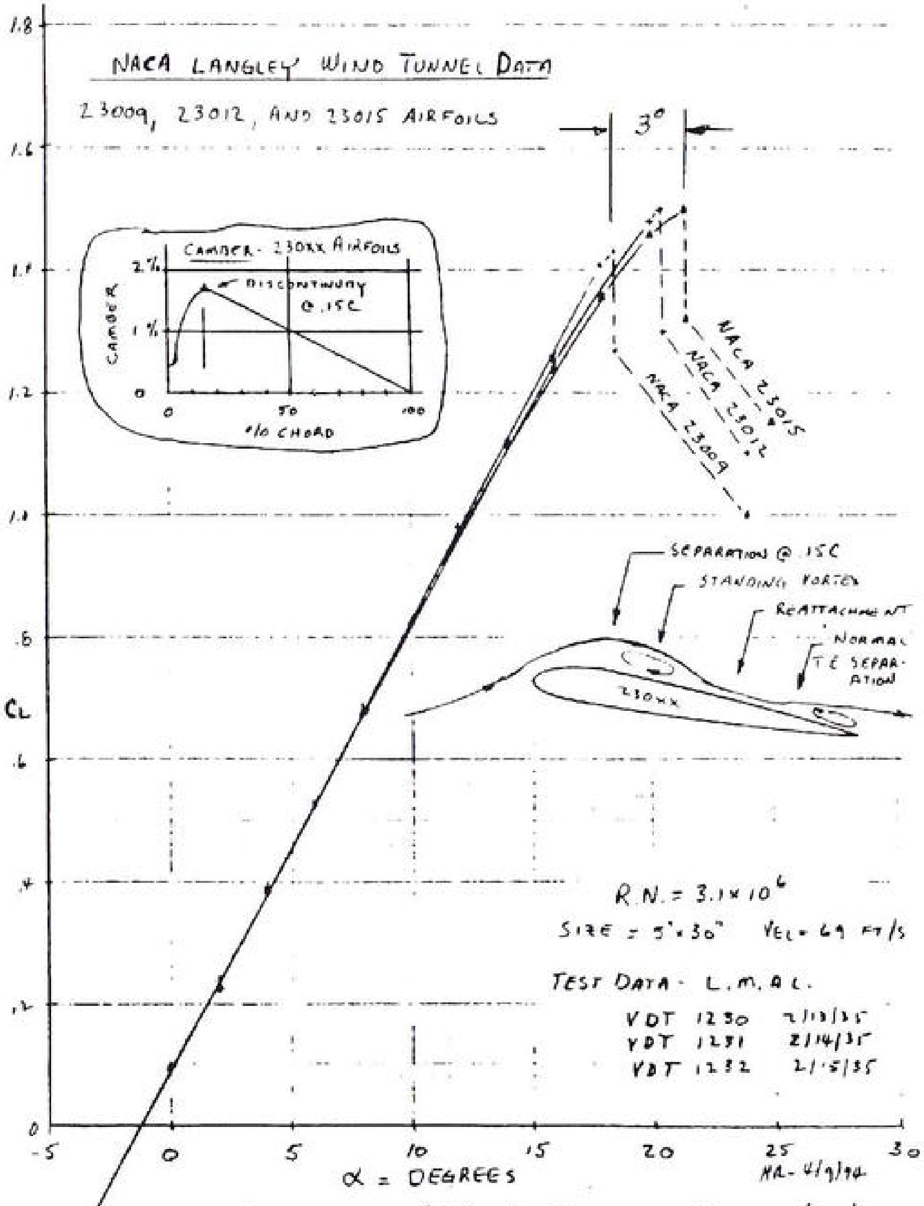
4

FIG 2+3



LIFT COEFFICIENT (C_L) Vs ANGLE OF ATTACK (α)

FIGURE 2



LIFT COEFFICIENT (C_L) Vs ANGLE OF ATTACK (α)

FIGURE 3

Airfoil Applications

Airplane		Root	Tip
Military			
Lockheed	P-38 Lightning	23016	4412
Bell	P-39 Aircobra	0015	23009
Curtis	P-40 Warhawk	2215	2209
North American	P-51 Mustang	63-215 plus strake	66-212
Chance Vought	F4U Corsair	23018	23009
Boeing	B-17 Flying Fortress	0018	0010
Consolidated	B-24 Liberator	Davis	Davis
Curtis-Wright	C-46	23017	4410.5
Douglas	C-47, DC-3	2215	4412
Commercial			
Culver	Cadet	3414	3408 plus slots
Ryan	Navion	4415	6410
Beech	Model 35 Bonanza	23016.5	23012
Beech	Skipper	GAW-1	
Cessna	210	64A215(a=0.5)	64A412(a=0.5)
Cessna	310	23018	23009
Cessna	337 Skymaster	2412	2409
Cessna	402	23018	23009
Cessna	500 Citation I	23014	23012
Piper	PA-24 Commanche	64A215	Same, no twist
Piper	PA-28 Cherokee	65-415	
Piper	PA-38 Tomahawk	GAW-1	
Mitsubishi	MU-2	64A415	63A212
Grumman	Goose	23015	23009
Homebuilts			
Falco	F-8	64-212	64-210
Questair	Venture (from Malibu)	23017	23010
Lake	Buccaneer	4415	4409
Bushby	Midget Mustang	64A212	64A210
Wittman	Tailwind	4309 top,0006 bottom	
Glassair		GAW-2 Modified	
Lancair, Wheeler Express, Prescott Pusher		NLF(1)-0215F Modified	
Pulsar		MS(1)-0313	
Jones	White Lightning	66-215	66-215
Thorp	T-18	63-412 Modified	

FIGURE 4

CESSNA 310 VORTEX GENERATORS

NACA 23018 ROOT, NACA 23009 TIP

Boundary Layer Research and The Twin Cessna Flyers have teamed up to bring early model Cessna 310 owners the advantages vortex generators afford. After Boundary Layer Research (BLR) developed the kits for the Cessna 310 through the 310F, BLR and The Twin Cessna Flyers came to an agreement granting The Twin Cessna Flyers the exclusive marketing rights for the kits. According to Bob Desroche, President of BLR, the relationship is "a perfect match" in that it allows BLR to concentrate on the development of new products, while it places the marketing efforts with an organization that has direct contact with over 7000 twin Cessna operators.

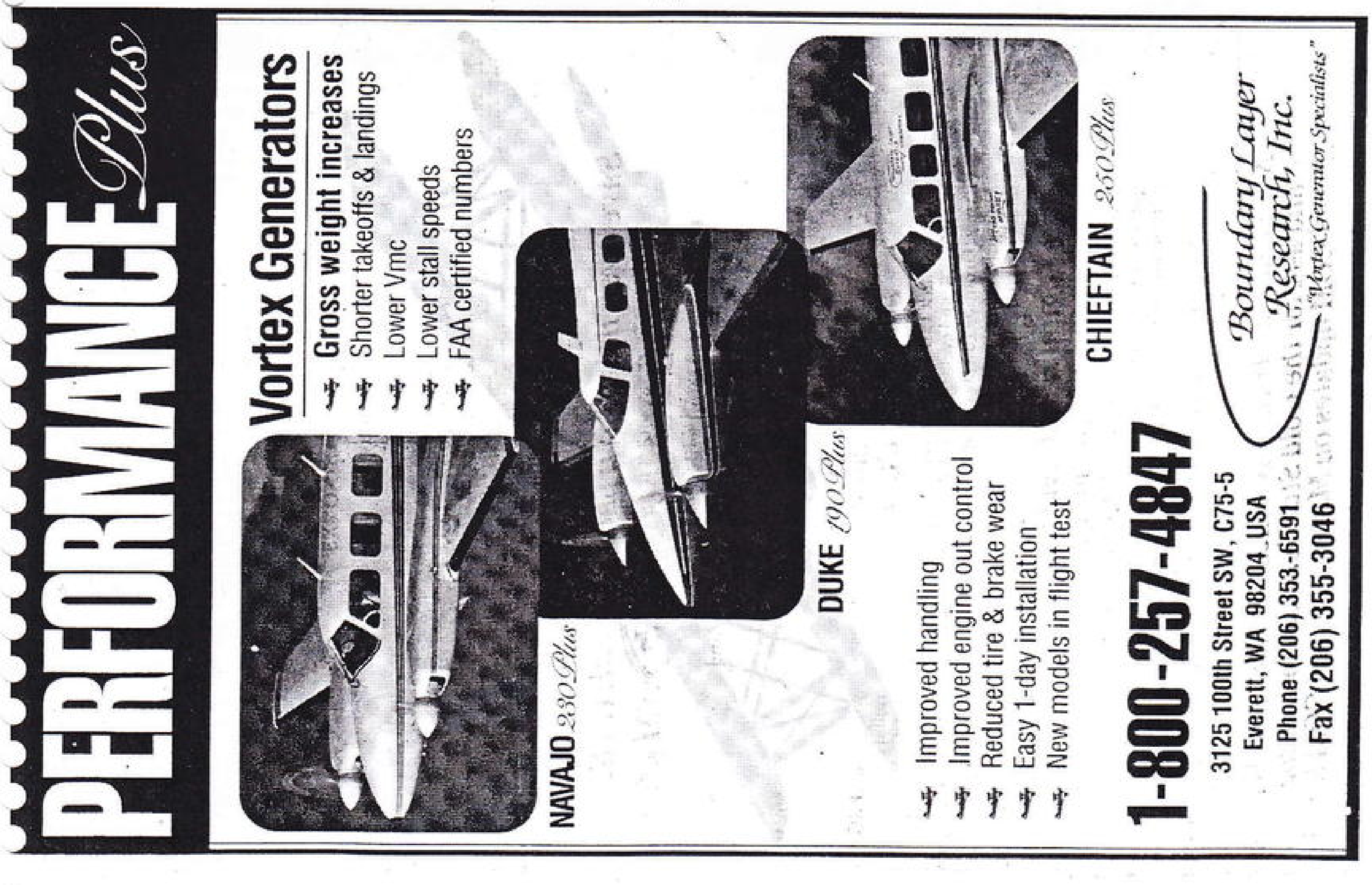
Installation of the kit can reduce Vmc by as much as 20 mph, with stall speeds reduced up to 8 mph, performance varies by model. Handling characteristics are dramatically improved. Low speeds control response remains positive, the tendency for the controls to become mushy is eliminated. In fact, VG equipped aircraft maintain full aileron control response all the way through a stall.

The kits consist of a series of small aluminum extrusions that are attached to the upper leading edge of the wings, up both sides of the vertical fin, and there are two strakes which are attached on the outboard side of the engine nacelles. Over simplified, the vortex generators "re-energize" the airflow over the control surfaces at low speeds and high angles of attack, and when the aircraft yaws, which increases the angle of attack of the vertical fin. This "re-energizing" causes the airflow to remain attached thus the stall is delayed and actually becomes much more docile when it does occur. Amazingly, this is all done with a kit that adds less than one pound to the weight of the aircraft.

Kits are available for the Cessna: 310, 310A, 310B, 310C, 310D, 310E, and 310F. For more information regarding these kits contact: The Twin Cessna Flyers at (800) 825-5310.

AIR PROGRESS

AUGUST 1994 27



Vortex Generators

- Gross weight increases
- Shorter takeoffs & landings
- Lower Vmc
- Lower stall speeds
- FAA certified numbers

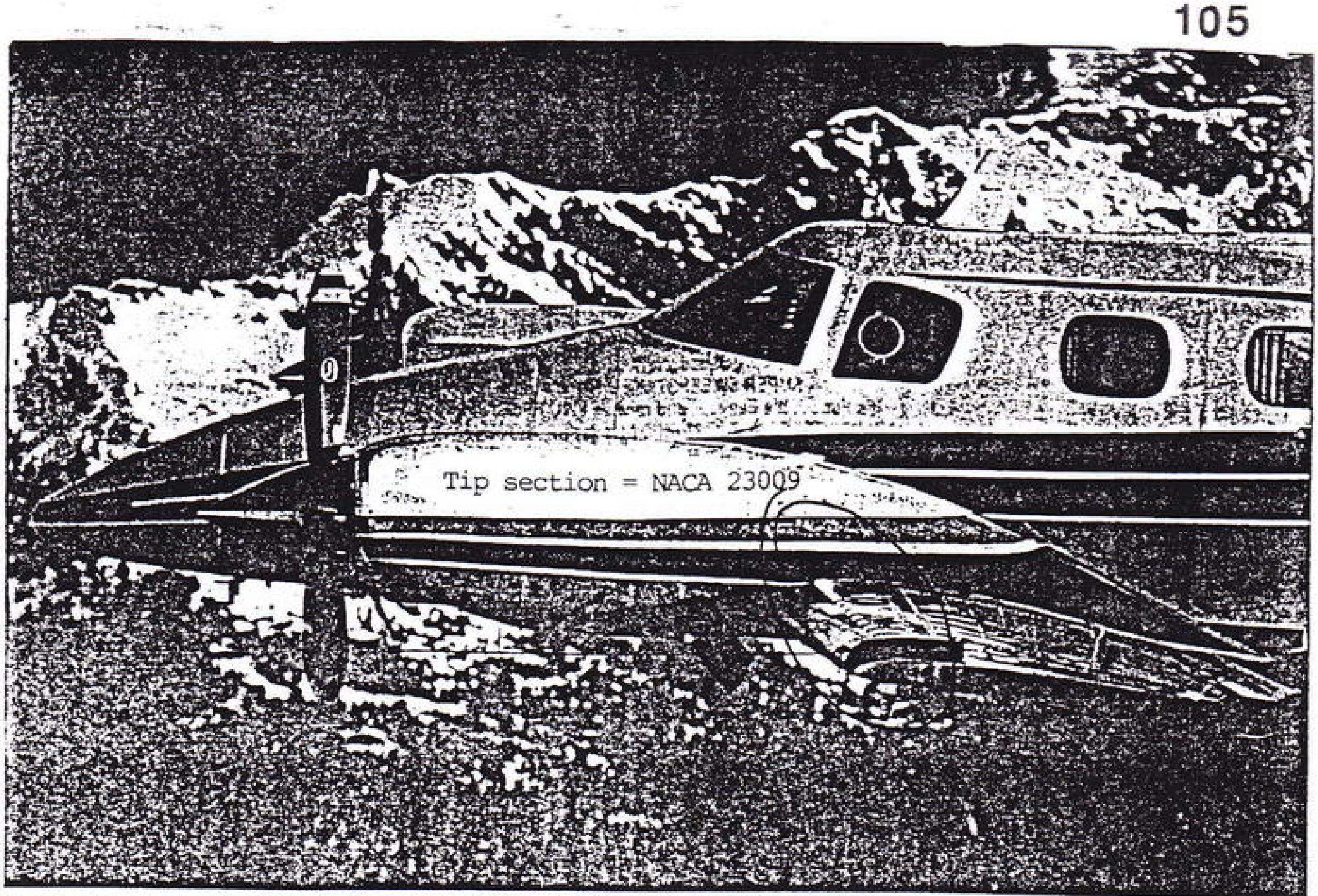
- Improved handling
- Improved engine out control
- Reduced tire & brake wear
- Easy 1-day installation
- New models in flight test

1-800-257-4847

3125 100th Street SW, C75-5
Everett, WA 98204 USA
Phone (206) 353-6591
Fax (206) 355-3046

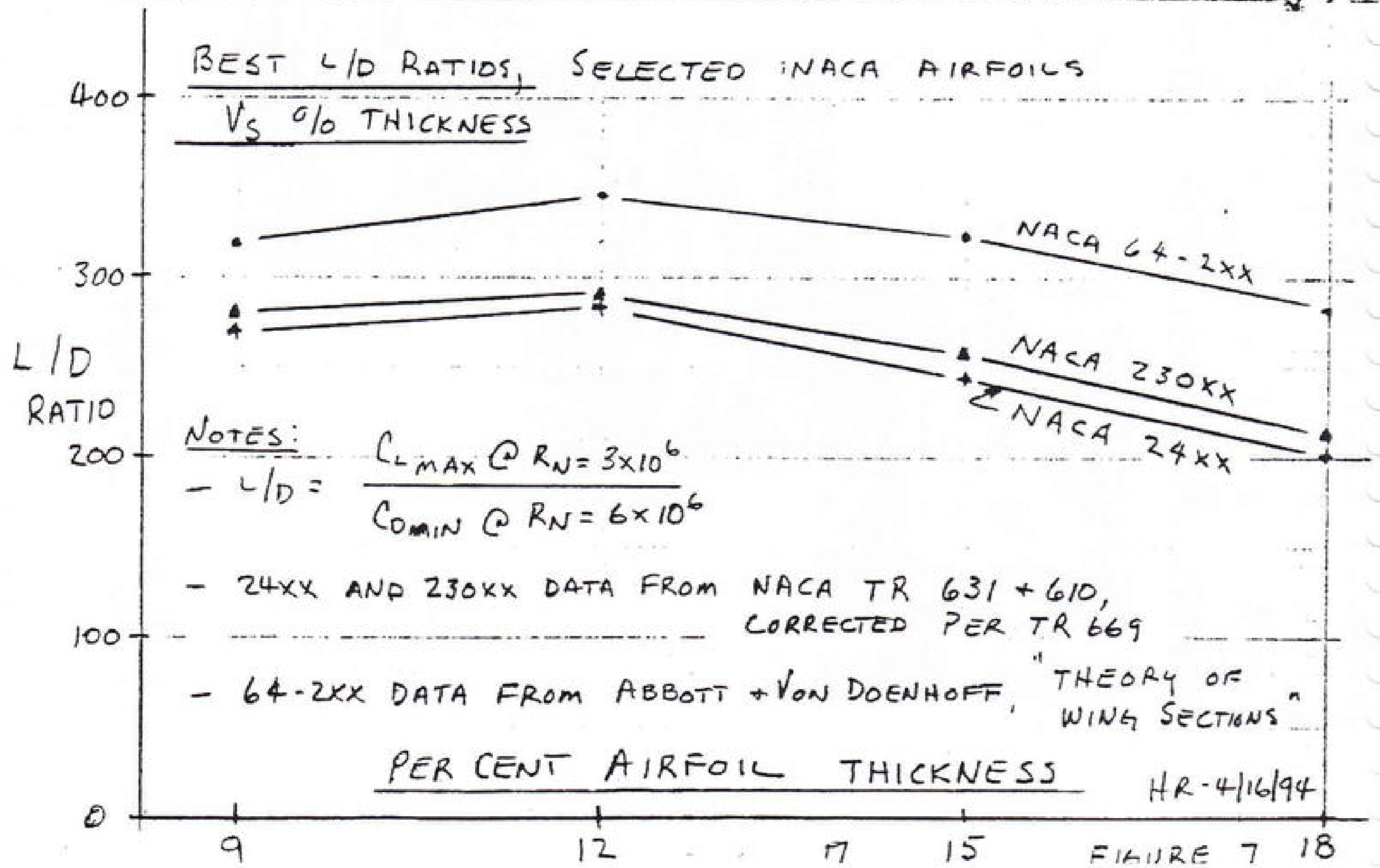
Boundary Layer Research, Inc.
"Vortex Generator Specialists"

FIGURE 6



The British Civil Aviation Authority has issued approval to Boundary Layer Research for installation of vortex generators on Beech Duke series 60, A60 and B60. The installation of the kit allows an increase of 190 pounds in gross takeoff weight. Gear extended stall speed is reduced from 74 knots to 69 knots. Vmc is virtually eliminated. For further information, contact BLR at (800) 257-4847, or fax (206) 355-3046.

"Air Progress" - Feb. 1994



BEST L/D RATIOS, SELECTED NACA AIRFOILS VS % THICKNESS

NOTES:

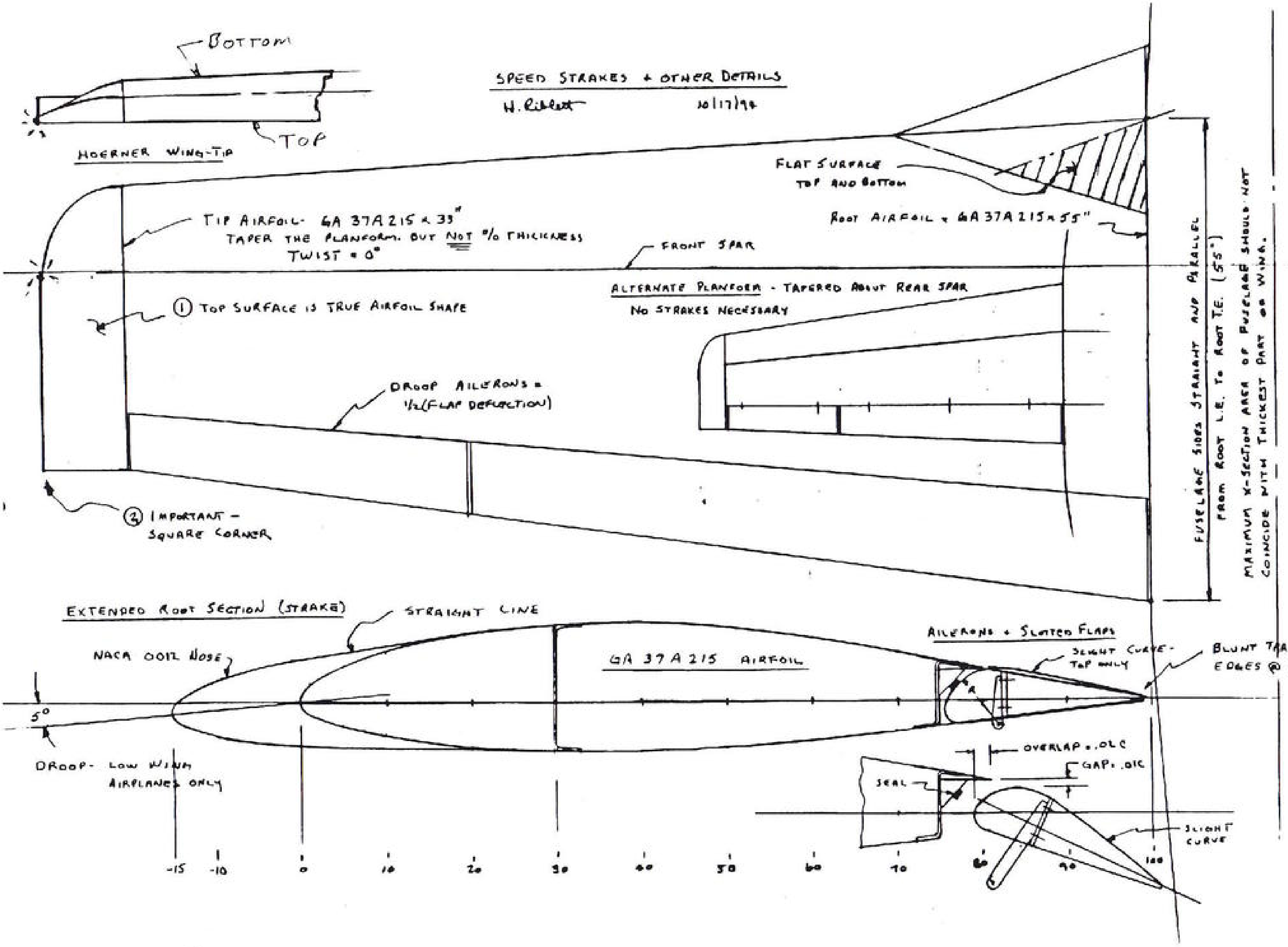
- $L/D = \frac{C_{L_{MAX}} @ R_N = 3\times10^6}{C_{D_{MIN}} @ R_N = 6\times10^6}$
- 24XX AND 230XX DATA FROM NACA TR 631 + 610, CORRECTED PER TR 669
- 64-2XX DATA FROM ABBOTT + VON DOENHOFF, "THEORY OF WING SECTIONS"

HR-4/16/94

FIGURE 7

SPEED STRAKES + OTHER DETAILS

H. Riblett 10/17/94


BOTTOM
TOP
HOERNER WING-TIP
TIP AIRFOIL- GA 37A215 x 33"
TAPER THE PLANFORM. BUT NOT % THICKNESS
TWIST = 0°
① TOP SURFACE IS TRUE AIRFOIL SHAPE
② IMPORTANT - SQUARE CORNER
DROOP AILERONS = ½(FLAP DEFLECTION)
FRONT SPAR
FLAT SURFACE TOP AND BOTTOM
ROOT AIRFOIL = GA 37A215 x 55"
ALTERNATE PLANFORM - TAPERED ABOUT REAR SPAR
NO STRAKES NECESSARY
FUSELAGE SIDES STRAIGHT AND PARALLEL
FROM ROOT L.E. TO ROOT T.E. (55")
MAXIMUM X-SECTION AREA OF FUSELAGE SHOULD NOT COINCIDE WITH THICKEST PART OF WING.
EXTENDED ROOT SECTION (STRAKE)
STRAIGHT LINE
NACA 0012 NOSE
GA 37A215 AIRFOIL
AILERONS + SLOTTED FLAPS
SLIGHT CURVE - TOP ONLY
BLUNT TRAILING EDGES @ .99C
R
5°
DROOP - LOW WING AIRPLANES ONLY
OVERLAP = .01C
GAP = .01C
SEAL
SLIGHT CURVE
-15
-10
0
10
20
30
40
50
60
70
80
90
100

Harry Riblett
416 Riblett Lane
Wilmington, DE 19808
302/994-0479

ADDENDUM NO. 3

(10/31/1995) Taming the Sharp Stall of the NACA 23012 Airfoil

Summary. This paper discusses two methods that can be used to soften the notoriously sharp stall of the NACA 23012 airfoil, that is, use of vortex generators, or alternatively an upper surface build-up (reprofiling). We also discuss the mechanism of the stall on this airfoil, and explain why additional leading edge droop is ineffective for improving the stall on this airfoil.

Discussion. In the mid-1930's, following the work on the 4-digit airfoil series, NACA conducted a series of tests on airfoils related to the 4-digit airfoils (same thickness distribution), but with the maximum camber placed unusually far forward. The object of the tests was to see if the airfoil Cm could be reduced to zero, which is not a necessity for conventional airplanes anyway, as long as the Cm is kept reasonably low by moderately forward loading. Anyhow, these "zero Cm" airfoils were designed and tested, and the most widely used of these 5-digit airfoils is the ubiquitous and infamous NACA 23012 airfoil. It is 12% thick, with peak camber of 1.8%C located at the 15%C position, rather than the usual 40%C position used on most of the popular 4-digit airfoils (2412, 4412, 4415, etc.). The mean line aft of .15C is a straight line, thus the only camber in this airfoil is in the first 15% of the chord length. Accordingly, we can describe the airfoil as being the NACA 0012 symmetrical section with the first 15% bent downwards (leading edge droop) approximately 1.8%. Actually, the effective droop is only about 1.4% due to the faulty "slope and radius" method of leading edge design used on all of the NACA airfoils, however the fact remains that the camber profile of this airfoil consists of leading edge droop only, with no conventional camber. The result of this is an airfoil with near-zero pitching moment coefficient (Cm). Further, since there is no negative camber in the airfoil, the maximum lift coefficient remains high and the induced drag is low, compared to the best known previous zero-Cm airfoil, Max Munk's 1924 M-6. The M-6 achieved zero Cm by reflexing (negative camber) the mean line from .60C to the trailing edge, effectively killing the nose-down pitching moment by applying a download on the trailing edge of the airfoil (figure 9). This, however increases total induced drag, and reduces Clmax. The camber profile of the M-6 in the first 60%C is conventional, however, so the M-6 does have a nice, soft stall.

Wind tunnel test results of the "new" zero-Cm airfoil, 23012, are summarized in NACA TR #537 of May 7, 1935 (see figure 1). Based on the fact that 23012 showed a moderately high Clmax, very low Cm, and Cd no greater than the 0012 symmetrical section, NACA pronounced this airfoil to be "markedly superior to well-known and commonly used sections", and recommended its wide usage in glowing terms. However, the airfoil has a terrible sharp-stall characteristic, which NACA TR#537 failed to discuss, and that is its downfall. This was noted briefly in Table II Airfoil Data of TR #537, so NACA knew of the existence of the sharp stall, and chose to ignore it. Sharp stall airfoils are bad enough on single engine airplanes, but on prop driven twins with one engine out these airfoils are especially lethal, causing accidents such as described in figure 2. In fact, sharp stall airfoils are the major reason that GA light twin fatality rates ironically exceed the fatality rates of GA light singles. We will not achieve true twin-engine reliability in prop-driven twins until we get rid of these sharp-stall airfoils. Prop-driven twins are limited by lateral control authority near Vmc due to the effect of a "blown surface" aft of the operating engine. With sharp-stall airfoils, sudden uncommanded and uncontrollable upsets occur at relatively high Vmc. With soft-stall airfoils, Vmc is much lower, and in addition the roll tendency is controllable. Knowing what we know today, we realize that NACA, as soon as they learned of the bad stall characteristic of the 5-digit airfoils, should have terminated the project. Accepting the sharp stall merely to achieve zero Cm was a poor trade-off.

There are two types of airfoils that have sharp stalls, those with too little camber in the leading edge, and those with too much camber in the leading edge. Examples of the first group include most symmetrical sections, and low-cambered airfoils such as the later NACA 64-212. These airfoils experience complete and sudden flow separation from the very leading edge at the stall, and they can usually be improved by adding a small amount of leading edge droop. Airfoils of the second group, including 23012, experience complete flow separation on the top surface at the stall, from a point near the end of the leading edge droop, that is, at about .12C in the case of the 23012. The result is the same, however- a sharp loss of lift, usually accompanied by loss of lateral control, and a hysteresis loop requiring a substantial decrease of angle of attack (with considerable altitude loss) before attached flow can be re-established.

Airfoils of the second group cannot be improved by adding leading edge droop, since they already have too much droop. For example, adding more droop to the 23012 airfoil results in the 33012 or 43012 airfoils, and these have stall characteristics as bad as or worse than the 23012, from wind tunnel data (check it out). What is required is to ease the transition from the leading edge droop to the rest of the mean line, reducing the discontinuity in the mean line at that point. One effective and proven method is to install a complete span-wise array of vortex generators on the top surface at about .10C. (Fig. 3, deleted). These function by filling in the "low spot" on the wing downstream of the VG's with a thickened boundary layer of energized turbulent air, discouraging flow separation. This fix is cheap and effective, and should be required on all twin-engine prop commuters using 5-digit airfoils, which is the majority of the fleet. Also, don't forget that that single engine airplanes with 5-digit airfoils can benefit from this as well.

Another possible way of accomplishing the same fix is to reprofile the wings, filling in the low spot aft of the leading edge droop on the top surface with solid material such as foam and glass. This is quite common on experimental (homebuilt) airplanes, with both metal and composite wings. Figures 4, 5, and 6 describe this method. Notice that the section drag actually decreases, in spite of the increased section thickness, due to the promotion of laminar flow. Further, the zero-lift Cm remains about the same, so top speed is unaffected. This new profile could also be used to modify existing tooling on production airplanes. Of course, the better solution aerodynamically is to discard the 23012 airfoil completely, and use a modern low-Cm, soft-stall airfoil, such as a "GA" airfoil.

As stated above, the NACA 5-digit sections cannot be improved by adding leading edge droop, thus the NC State/NASA fix on the "Venture" airplane (figure 8) is a poor solution to the problem. I suspect that any improvement of the stall in this case is merely the result of the considerable aerodynamic twist that was introduced into the wing by this fix, delaying the tip stall. However, this causes a considerable loss of efficiency at high speed, and also raises the landing speed of the airplane, so it is not a good solution to the problem.

4301X Airfoils. The NACA 4301X airfoils are the same as the 2301X airfoils, but with twice as much leading edge droop (figure 9). The stall is as bad as, or worse than, the 2301X airfoils. Fortunately, the 4301X airfoils are rarely used. One exception is Fred Weike's "Ercoupe" (NACA 43013), however on this airplane the elevator travel is limited so that the stall angle of attack can never be reached. Still, the 43013 was a strange choice for this airplane.

Another airplane that uses the 4301X airfoils is the French ATR-72 twin turboprop commuter, and that story does not have such a happy ending. In October 1994 a crash occurred at Roselawn, Indiana killing all 68 people on board, a direct result of the NACA 4301X airfoils used on that airplane (figure 10). As confirmed by subsequent in-flight spray tests, a 3/4 inch high ice ridge forms on the top surface at .09C, forming a very effective spoiler. When combined with the already terrible stall characteristic of the airfoil, this caused an uncommanded, uncontrollable roll-over and irrecoverable dive. The ice ridge forms at the very spot on the airfoil where a pronounced suction peak exists (figure 7), since here the temperature depression from adiabatic expansion and cooling is the greatest. The ice ridge is aggravated by the strange placement of the vortex generators on the ATR 72 wing. The "fix", longer de-icer boots, is merely a bandage, for the basic problem remains. Modern constant-velocity airfoils do not have this suction peak, nor the problem.

Conculsions. These interim fixes should be used until the NACA 5-digit airfoils can design calculations. The simple mean-line airfoils gave exceptionally low pitching moments, somewhat lower than the theoretical values based on the mean line. Both the measured and theoretical curves for the simple mean-line airfoils are given in figure 16.

The analysis of these charts and the data of table II show that the reflexed airfoils, although comparing favorably with other reflexed airfoils, are surpassed by the simple mean-line airfoils. Furthermore, the airfoils covering a range of camber locations forward of normal positions possess improved characteristics.

A comparison of the N. A. C. A. 24012 with the N. A. C. A. 2212 indicates the differences that may be attributed to the difference between the mean-line forms. These airfoils having the same camber location but different mean-line forms possess approximately the same lift and drag characteristics. The angle of zero lift and the pitching moment, however, are quite different. Especially noteworthy is the very much lower pitching moment produced by the airfoils reported herein.

One of the promising airfoils of this group, the N. A. C. A. 23012 airfoil (previously referred to as "the N. A. C. A. A-312"), has been further investigated by tests in the full-scale tunnel and over a range of values of the Reynolds Number in the variable-density tunnel. These results (reference 3) confirm the conclusion that this airfoil has improved characteristics over well-known and commonly used airfoils of this class. It has a high maximum lift and a low pitching moment. Furthermore, the minimum drag is practically as low as that of the corresponding symmetrical airfoil, the N. A. C. A. 0012.

More generally, other sections of this group, such as the N. A. C. A. 21012 and 22012 having an even lower pitching moment than the 23012, should supply the need of many applications requiring a slightly cambered section of moderate thickness having a very low pitching moment.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., *May 7, 1935.*

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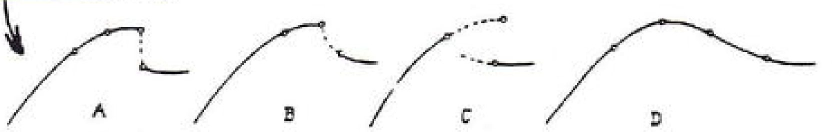
1. Jacobs, Eastman N., Ward, Kenneth E., and Pinkerton, Robert M.: The Characteristics of 78 Related Airfoil Sections from Tests in the Variable-Density Wind Tunnel. T. R. No. 460, N. A. C. A., 1933.
2. Jacobs, Eastman N., and Abbott, Ira, H.: The N. A. C. A. Variable-Density Wind Tunnel. T. R. No. 416, N. A. C. A., 1932.
3. Jacobs, Eastman N., and Clay, William C.: Characteristics of the N. A. C. A. 23012 Airfoil from Tests in the Full-Scale and Variable-Density Tunnels. T. R. No. 530, N. A. C. A., 1935.

TABLE II.—AIRFOIL DATA

Airfoil	Classification				Effective R. N., millions [5]	$C_{L_{max}}$	α_0 at $C_{L_{max}}$ (degrees)	Fundamental section characteristics							Derived and additional characteristics that may be used for structural design								
													a. c. (percent c from c/4)			c. p. at $C_{L_{max}}$ (percent c)		Wing characteristics A. R. 6 rounded tips		Thickness (percent c) at—			
	Chord [1]	P.D. [2]	S.E. [3]	$C_{L_{max}}$ [4]				α_{l_0} (degrees)	a_0 (per deg.)	$c_{l_{opt}}$	$c_{d_{0\,min}}$	$c_{m_{a.c.}}$	Ahead	Above	$\frac{C_{L_{max}}}{c_{d_{0\,min}}}$	Computed	Actual	m (per rad.)	$C_{D_{0\,min}}$	0.15c	0.60c	Maximum	Camber (percent c)
						(6)			(6)	(6)	(6)	(6)	(6)	(6)	(6)	(6)							
N. A. C. A. 21012	A	C12	D2	C	8.37	1.67	15	−0.6	0.099	0.00	0.0070	0.001	2.0	3	239	23	23	4.32	0.0070	10.69	8.26	12	1.1
N. A. C. A. 22012	A	C12	D2	C	8.32	1.66	16	−.9	.100	.10	.0071	−.005	1.8	3	234	24	23	4.34	.0073	10.69	8.24	12	1.5
N. A. C. A. 23012	A	C12	D2	A	8.48	1.67	16	−1.2	.100	.08	.0069	−.005	1.7	3	242	24	26	4.34	.0070	10.69	8.25	12	1.8
N. A. C. A. 24012	A	C12	C3	C	8.26	1.65	16	−1.5	.100	.08	.0072	−.013	1.7	3	229	24	26	4.34	.0073	10.71	8.25	12	2.1
N. A. C. A. 25012	A	C12	C3	C	8.24	1.61	15	−1.6	.100	.10	.0075	−.019	1.6	3	215	25	27	4.34	.0077	10.72	8.26	12	2.3
N. A. C. A. 0012	A	C10	C0	A	8.53	1.58	17	.0	.097	.00	.0068	0	1.6	2	232	23	25	4.25	.0066	10.69	8.27	12	0
N. A. C. A. 2212	A	C12	C3	B	8.42	1.65	16	−1.8	.099	.12	.0072	−.029	1.4	2	229	23	27	4.32	.0073	10.69	8.25	12	2.0
N. A. C. A. 22112	A	C11	D2	B	8.55	1.58	15	−.8	.100	.06	.0072	.001	1.5	3	219	23	25	4.34	.0073	10.69	8.27	12	1.6
N. A. C. A. 23112	A	C11	D2	A	8.21	1.67	16	−.6	.100	.08	.0074	.002	2.0	4	228	23	25	4.34	.0073	10.69	8.28	12	2.1
N. A. C. A. 24112	A	C11	C3	A	8.00	1.61	16	−.9	.100	.10	.0074	0	1.9	4	218	23	25	4.34	.0075	10.71	8.26	12	2.4
N. A. C. A. 25112	A	C11	C3	B	8.21	1.56	15	−1.2	.100	.08	.0074	−.002	1.6	3	211	23	27	4.34	.0075	10.73	8.25	12	2.7

[1] Type of chord. A refers to a chord defined as a line joining the extremities of the meanline.
[2] Type of pressure distribution.
[3] Type of scale effect on maximum lift.
[4] Type of lift-curve peak as shown in the sketches.



NOTE.—The foregoing classifications are given here for convenient future reference. A detailed discussion will be published in a later report dealing with the application of airfoil characteristics to design problems.

[5] Turbulence factor is 2.64.
[6] These data have been corrected for tip effect.

FIGURE 1

BEECH KING AIR ACCIDENT REPORT

Local

Monday, May 3, 1993 • Page A 3

The News Journal, Wilmington, Del.

LOTTERIES OBITUARIES WEATHER

Plane lost airspeed just before NCCo crash

By ERIC RUTH
Staff reporter

HARES CORNER — Its left engine ad and the runway a quarter mile vay, a crippled airplane dropped well low the minimum speed required for ght just before it smashed into a stand trees and burst into flames at New astle County Airport last June, killing l four aboard, investigators say.

But the newly released report to the ational Transportation Safety Board ye this type of jet-powered propeller ane, usually is capable of maintaining ght with one engine running — even uring landing. That leaves unresolved hy the airspeed fell lower as the Beech ing Air 200 made its way toward Runay 19.

Aviation experts told investigators, owever, that pilots confronted by a single-engine situation must react swiftly and recognize that certain steps are crucial to maintaining speed. Immediately after the engine fails, the pilot must push the running engine to full power, and reduce drag by turning the blades of the other propeller into the wind or air stream.

"If the pilot does not recognize the engine failure quickly enough or hesitates, the airspeed will decay and he could be in trouble," said Lester H. Berven, an engineering test pilot for the Federal Aviation Administration.

Tests confirmed that the left engine was not running at impact. It failed when a cracked blade of its whirling inner fans broke loose and tore up other parts. Investigators confirmed that the right engine was running, possibly at full power.

Pilots of twin-engine aircraft are taught to recognize that the plane will safely fly at a minimum speed only if the wings are banked toward the running engine. Flying with wings level will bring a loss of control at a significantly higher speed, experts said.

For example, in one type of aircraft the minimum single-engine speed is 105 mph. But without the wings banked, the plane actually will start to become uncontrollable at 132 mph, the report said.

Operating handbooks for the King Air state that when just one engine is running, pilots should maintain 162 mph, assuming wings are properly banked. The recommended approach airspeed for normal flight conditions is 151 mph.

Radar records showed that the airspeed of the King Air that crashed, call sign "N32HG," dropped from 145 mph to 135 mph as it approached the runway. A witness who is a commercial pilot reported that the wings appeared level at an altitude of about 400 feet.

When airspeed drops below the minimum point, the airplane tends to roll toward the dead engine, even more so as airspeed falls, the report said. "If [loss of lift] . . . should occur in this condition, a violent roll into the dead engine may be experienced," the FAA warns pilots in pamphlet. Directional control is even more difficult when, as in the case of N32HG, the left engine has failed.

Witnesses reported that N32HG rolled rapidly toward the dead engine just before its crash.

Killed in the crash were pilot James W. Ball, 56, of Baltimore; co-pilot James R. Kestner, 55, of Lititz, Pa.; Peter F. Osterchrist, 55, of Timonium, Md.; and his wife Elizabeth M. Osterchrist, 56.

The safety board will use the report to determine the accident's probable cause.

IS THIS 'PILOT ERROR', OR POOR AIRFOIL SELECTION?

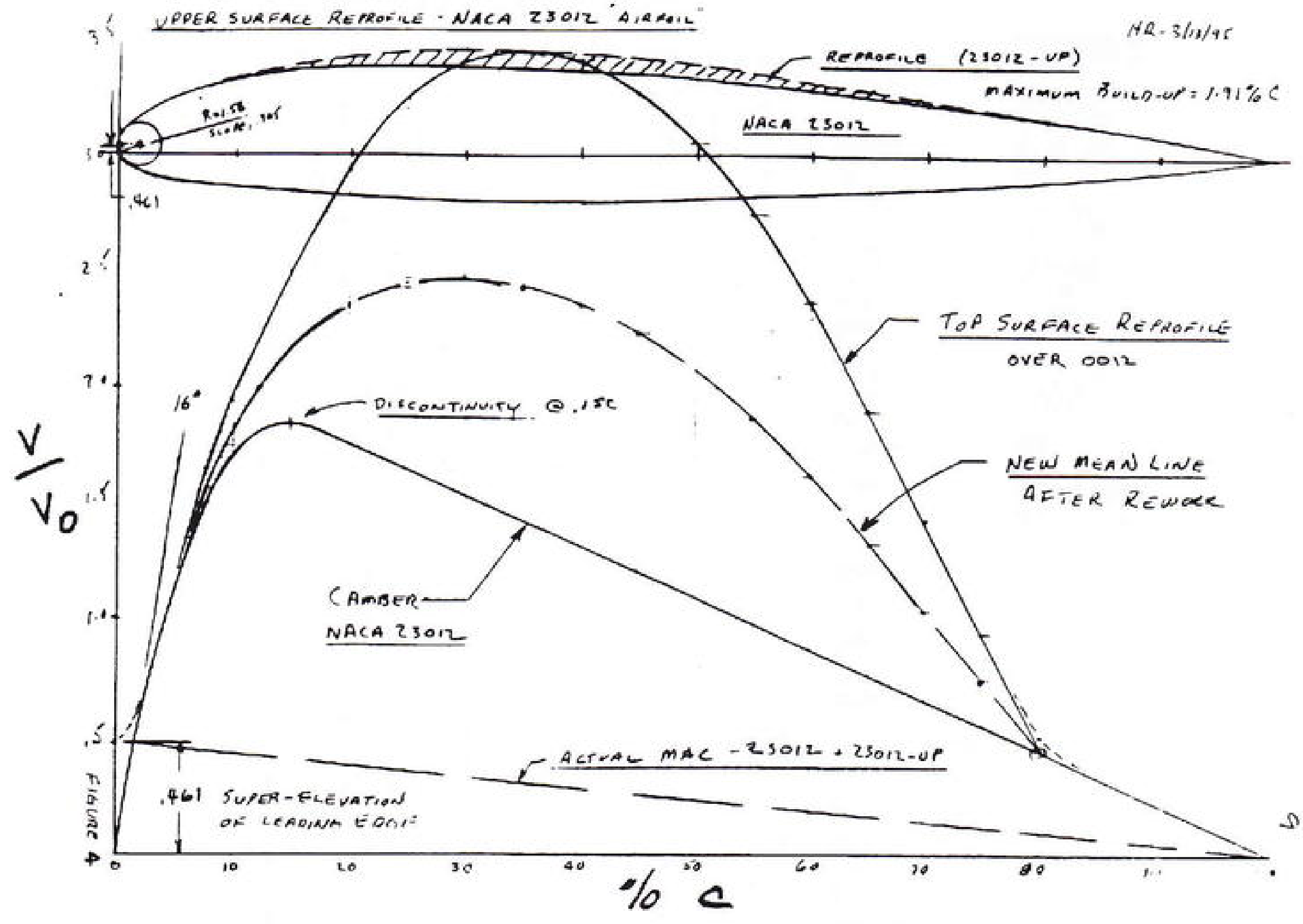
FIGURE 2

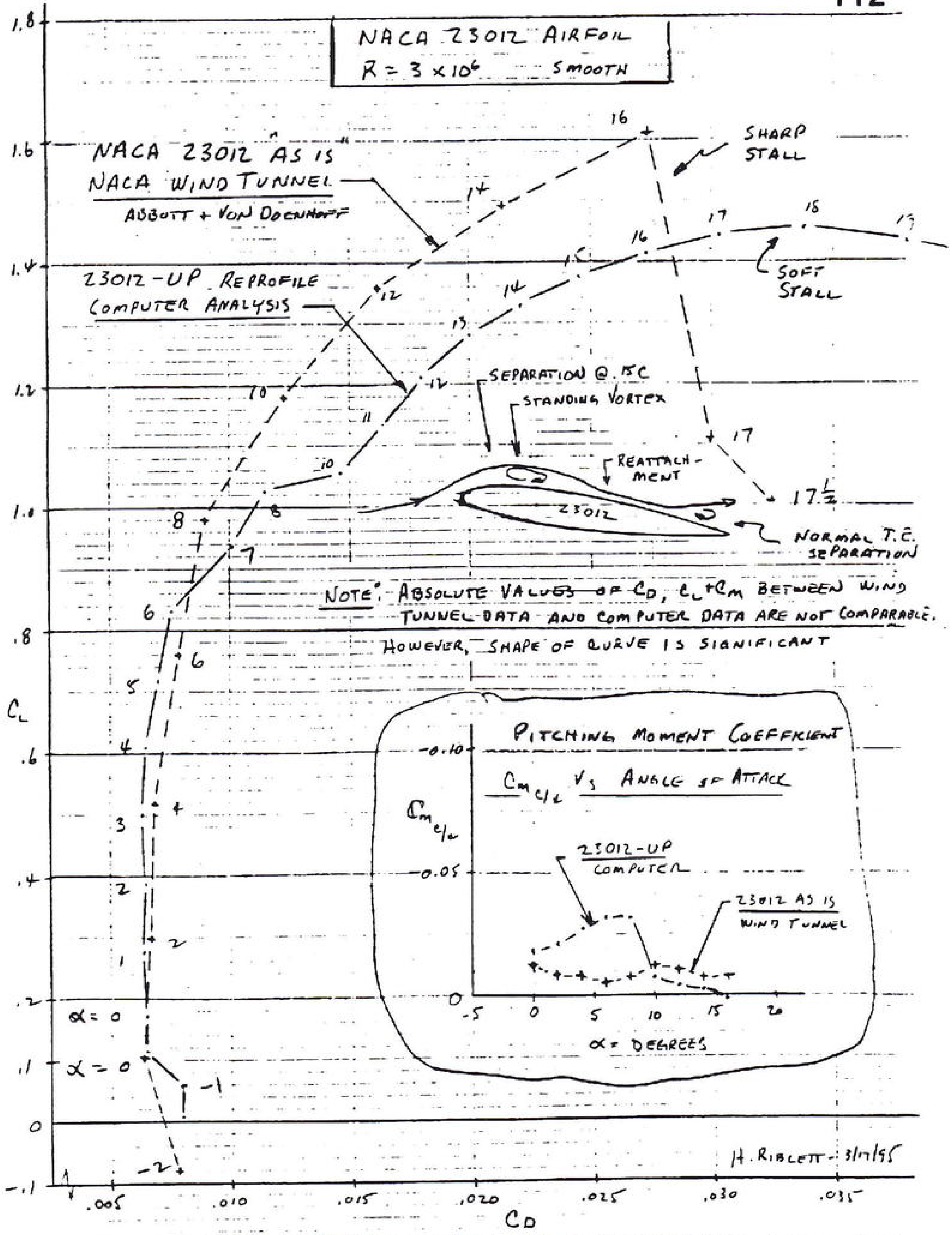
UPPER SURFACE REPROFILES - NACA 230XX AIRFOILS

HR-3/28/95

Mar 95 Sta	NACA 0012	Reprofile over 0012	23010-UP top ord	23012-UP top ord	23015-UP top ord	23017-UP top ord	23018-UP top ord
7.50	4.220	1.650	5.167	5.870	6.925	7.628	7.980
10.00	4.682	1.970	5.872	6.652	7.823	8.603	8.993
15.00	5.345	2.500	6.954	7.845	9.181	10.072	10.518
20.00	5.738	2.930	7.711	8.668	10.103	11.059	11.537
25.00	5.942	3.240	8.191	9.182	10.668	11.658	12.153
30.00	6.001	3.420	8.421	9.421	10.921	11.922	12.422
35.00	5.949	3.450	8.407	9.399	10.886	11.878	12.374
40.00	5.803	3.390	8.226	9.193	10.644	11.611	12.095
45.00	5.581	3.260	7.911	8.841	10.236	11.167	11.632
50.00	5.294	3.050	7.461	8.344	9.668	10.550	10.991
55.00	4.953	2.750	6.877	7.703	8.941	9.767	10.180
60.00	4.563	2.350	6.152	6.913	8.054	8.814	9.195
65.00	4.133	1.890	5.334	6.023	7.056	7.745	8.090
70.00	3.664	1.420	4.473	5.084	6.000	6.611	6.916
75.00	3.160	0.950	3.583	4.110	4.900	5.427	5.690

FIG. 4





NOTE: ABSOLUTE VALUES OF CD, CL + CM BETWEEN WIND TUNNEL DATA AND COMPUTER DATA ARE NOT COMPARABLE. HOWEVER, SHAPE OF CURVE IS SIGNIFICANT

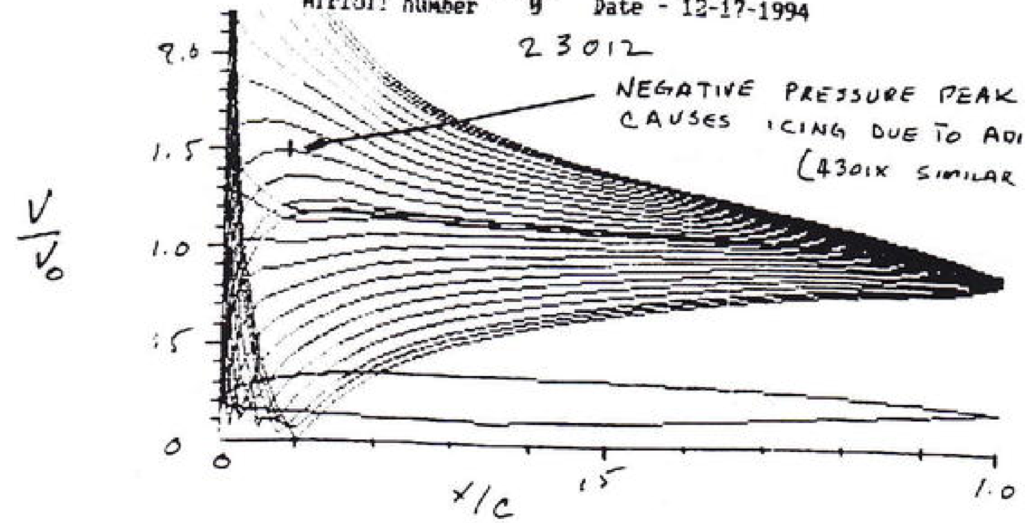
H. RIBLETT - 3/17/95

LIFT COEFFICIENT Vs DRAG COEFFICIENT FIGURE 6

NACA 23012 AIRFOIL - VELOCITY DIAGRAM

```
AIRFOIL 23012        [illegible] THICKNESS          .00% FLAP        .00 DEGREES DEFL
 11.00   13.00   15.00   [illegible]   [illegible]   [illegible]   3.00   5.00   7.00   9.
ELATIVE TO THE CHORD LINE VELOCITY DISTRIBUTIONS FOR THE ABOVE ANGLES OF ATT
 52   1.00000   .00000   .916   .917   .916   .914   .911   .907   .
 .896   [illegible]   [illegible]   .871   [illegible]   .865
```

Airfoil number 9 Date - 12-17-1994

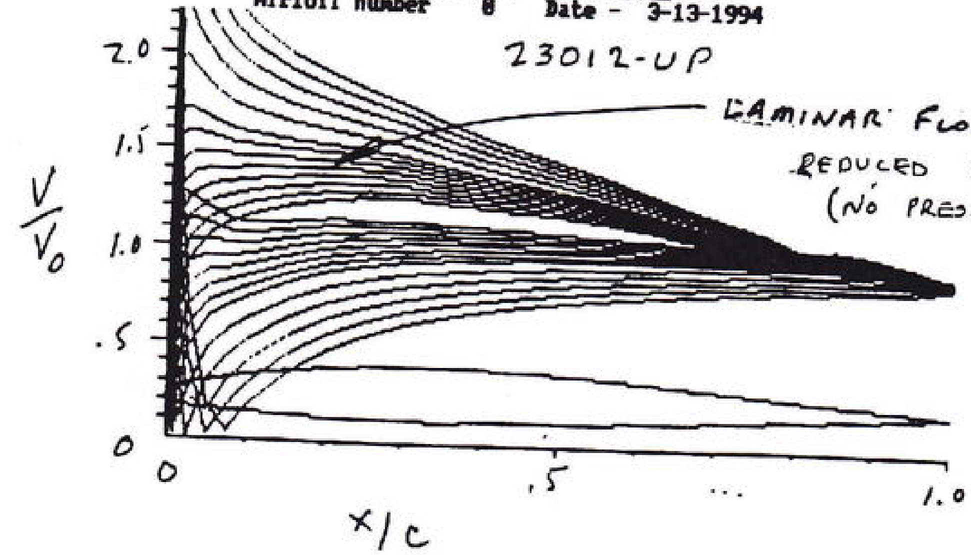


NEGATIVE PRESSURE PEAK @ .09C, α = 3° CAUSES ICING DUE TO ADIABATIC COOLING HERE (4301X SIMILAR BUT WORSE)

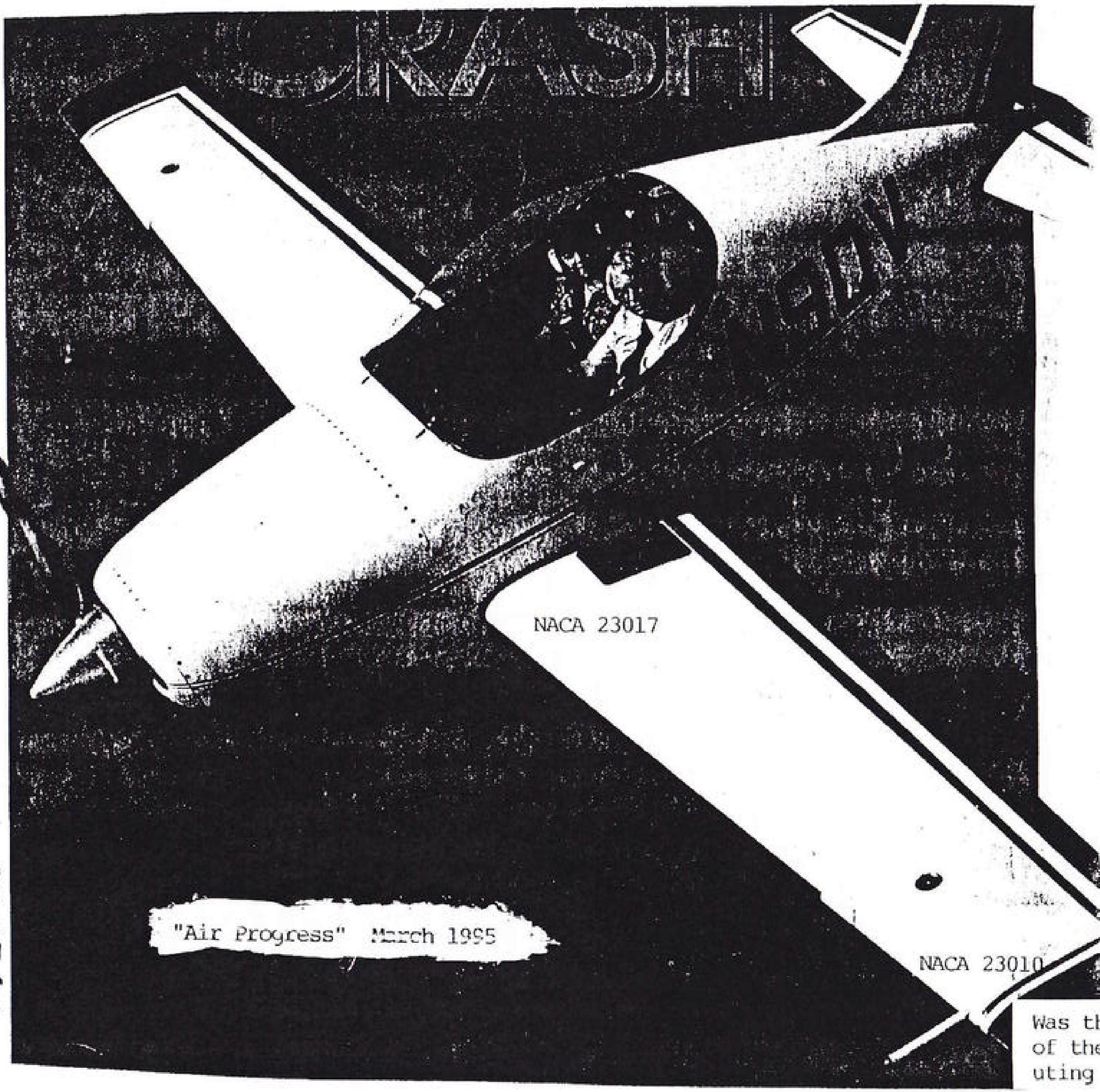
23012-UP AIRFOIL (REPROFILED UPPER SURFACE) VELOCITY DIAGRAM

```
AIRFOIL 12up                  13.93% THICKNESS          .00% FLAP        .00 DEGREES DEFLEC
ON
 6.00       8.00       10.00     -1.00       .00        1.00       2.00       3.00       4.00       5.00
 N          X          Y         12.00      14.00      16.00
ELATIVE TO THE CHORD LINE VELOCITY DISTRIBUTIONS FOR THE ABOVE ANGLES OF ATTACK
 52     1.00000     .00000      .914       .913       .913       .912       .911       .910       .908
 .906       .901       .896      .889       .881       .872
```

Airfoil number 8 Date - 3-13-1994



LAMINAR FLOW, TOP SURFACE, α=3° REDUCED ICING PROBLEM (NO PRESSURE PEAK)



"Air Progress" March 1995

FIGURE 8-1

SPORT AVIATION - MAR '95

Henry Bouley, 43, and **Robert W. "Bob" McLallen**, 60, died in the crash of Henry's Questair Venture on Saturday, January 21, 1995. According to the FAA preliminary report, the two were on an IFR flight from Chicago to Cheyenne, WY when they experienced an alternator failure near Des Moines, IA. Declaring an emergency, they were handed off to Des Moines Approach and began a circling descent from 15,000 feet. At approximately 3 miles southeast of the airport they reported that a dead stick landing was probable, and moments later the aircraft was observed spinning to the ground 3/4's of a mile from the end of Des Moines' Runway 31R. Weather was not a factor in the accident.

Bouley and McLallen were business associates in activities involving the Questair Venture. Henry was the designer and manufacturer of the PMA series of 6 cylinder aircraft engines and was to introduce a new 4 cylinder engine this spring. His operation at North Windham, CT was also the manufacturer of the Questair Venture and Spirit kit components and was a building center for the two designs. The winner of the 1993 Great Cross Country Air Race in the prototype Venture, Henry was featured in an article in **Sport Aviation** in the October 1993 issue.

Bob McLallen was the owner of the Questair Venture/Spirit kit business, having purchased it several years ago after the original company entered bankruptcy. He moved the firm from its original site in Greensboro, NC to Waukegan, IL last year and had contracted with Henry Bouley to manufacture the kit components. Bob was a resident of Northfield, IL and was the owner of an insurance company that specializes in coverage for apartments and condominiums. Both he and Henry Bouley were experienced, instrument rated pilots.

Was the power failure the true cause of the accident, or merely a contributing factor?

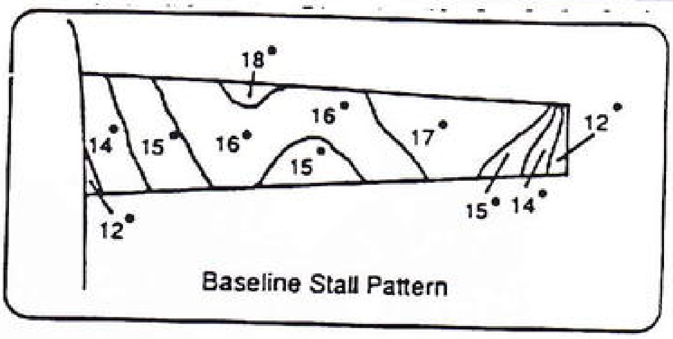
This is garbage

NORTH CAROLINA
STATE UNIVERSITY

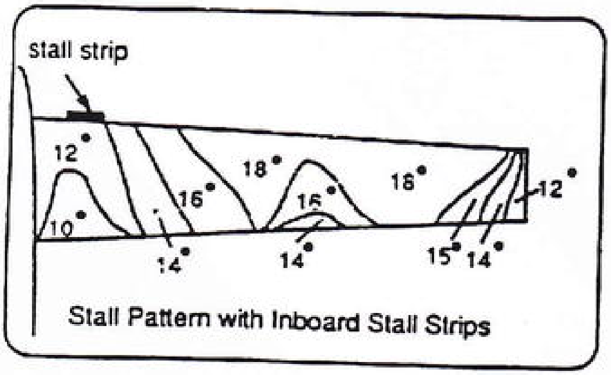
ENHANCED WING DESIGN FOR STALL DEPARTURE RESISTANCE

● THE HIGH ASPECT RATIO WING

While providing significant benefits in cruise and climb, the high aspect ratio wing requires more design consideration at stall due to the inconsistencies in the growth of its multiple stall cells.


18°
16°
14°
15°
16°
15°
17°
12°
12°
15°
14°

Baseline Stall Pattern


stall strip
12°
10°
16°
18°
16°
18°
12°
14°
14°
15°
14°

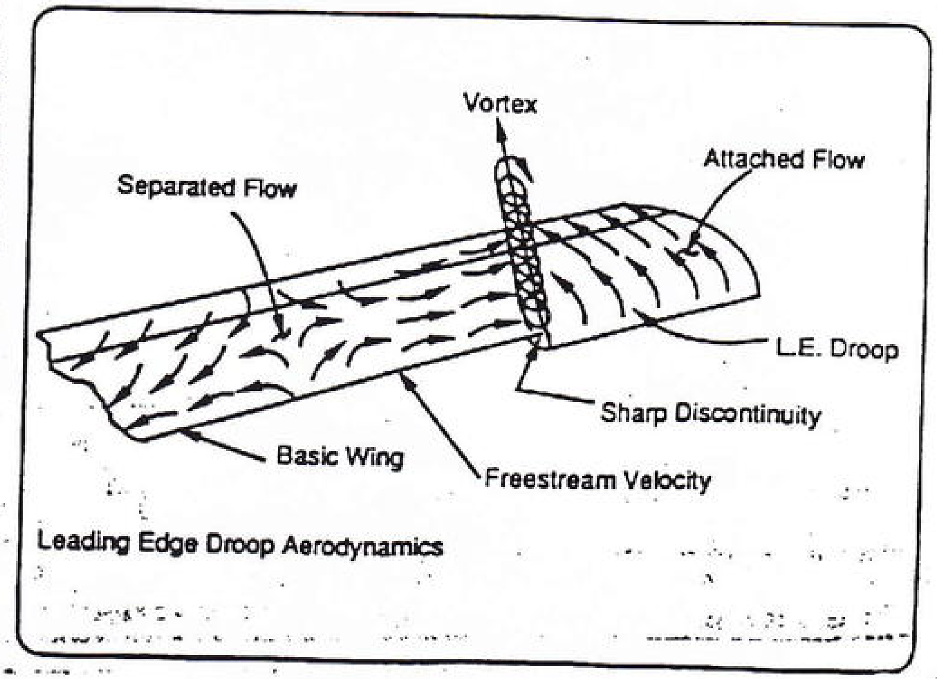
Stall Pattern with Inboard Stall Strips

● STALL STRIPS

Stall strips are widely used to force inboard separation before the stall cells fully develop. However, the resulting aircraft pitch-down limits the useable lift of the wing.

● THE LEADING EDGE DROOP

The design philosophy behind the NASA leading edge droop is to provide a "passive" device which delays the cutboard separation well beyond the point at which it would normally occur. This is achieved by the introduction of increased leading edge camber and the augmentation of the energy within the boundary layer created by a vortex which forms at the discontinuous wing/droop intersection. The net result is increased stall departure resistance due to the improved roll damping and predictable pitch break.

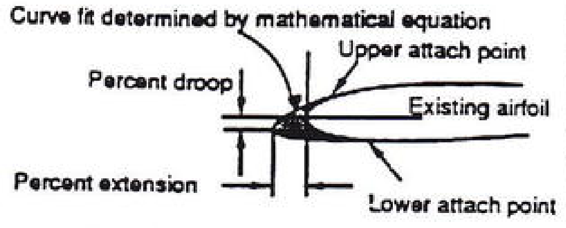

Vortex
Attached Flow
Separated Flow
L.E. Droop
Sharp Discontinuity
Basic Wing
Freestream Velocity

Leading Edge Droop Aerodynamics

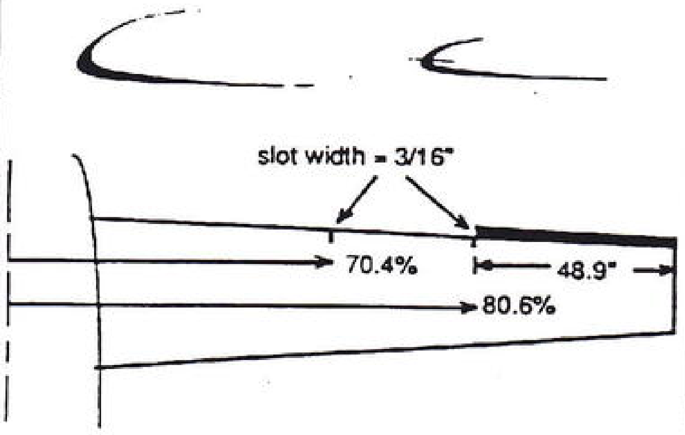
FIGURE 9.7

THE LEADING EDGE DROOP DESIGN PROCESS

Since leading edge droops are retro-fitted, the design process must accout for the aerodynamics of the existing wing. A computer code was developed to mathematically define, iterate, and optimize a leading edge extension of the current airfoil section. Minimizing the cruise-drag penalty while increasing the stall angle of attack are the primary design considerations.


Curve fit determined by mathematical equation
Upper attach point
Percent droop
Existing airfoil
Percent extension
Lower attach point

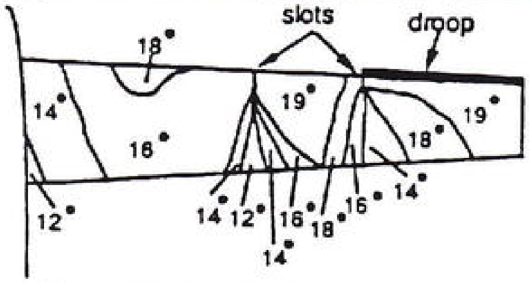
Leading Edge Droop Design Description


slot width = 3/16"
70.4%
48.9"
80.6%

Questair Venture Droop/Slot Design

THE VENTURE DROOP/SLOT DESIGN

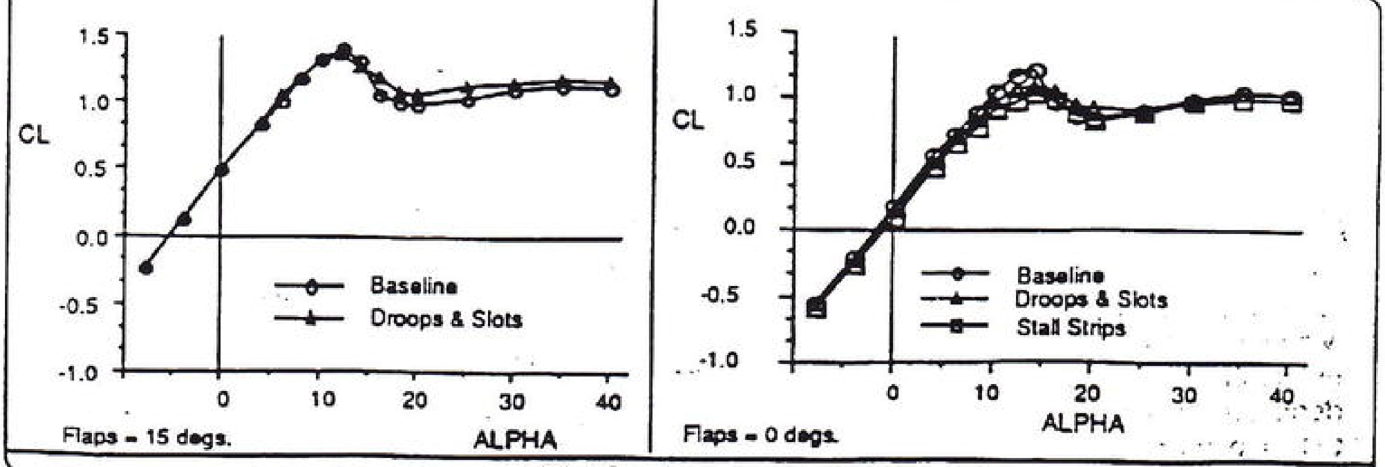
Designed at North Carolina State University and wind tunnel tested at NASA Langley, the Questair Venture modifications represent a unique blending of aerodynamic concepts. A conservative droop is coupled with a pair of chordwise leading edge slots. The slots generate additional vortices which act to prevent the progression of the primary stall cell. Thus, the combination of slots and droops allow more of the wing to attain attached flow to higher angles of attack.


slots
droop
18°
14°
16°
19°
18°
19°
12°
14
12°
16°
18
16°
14°
14°

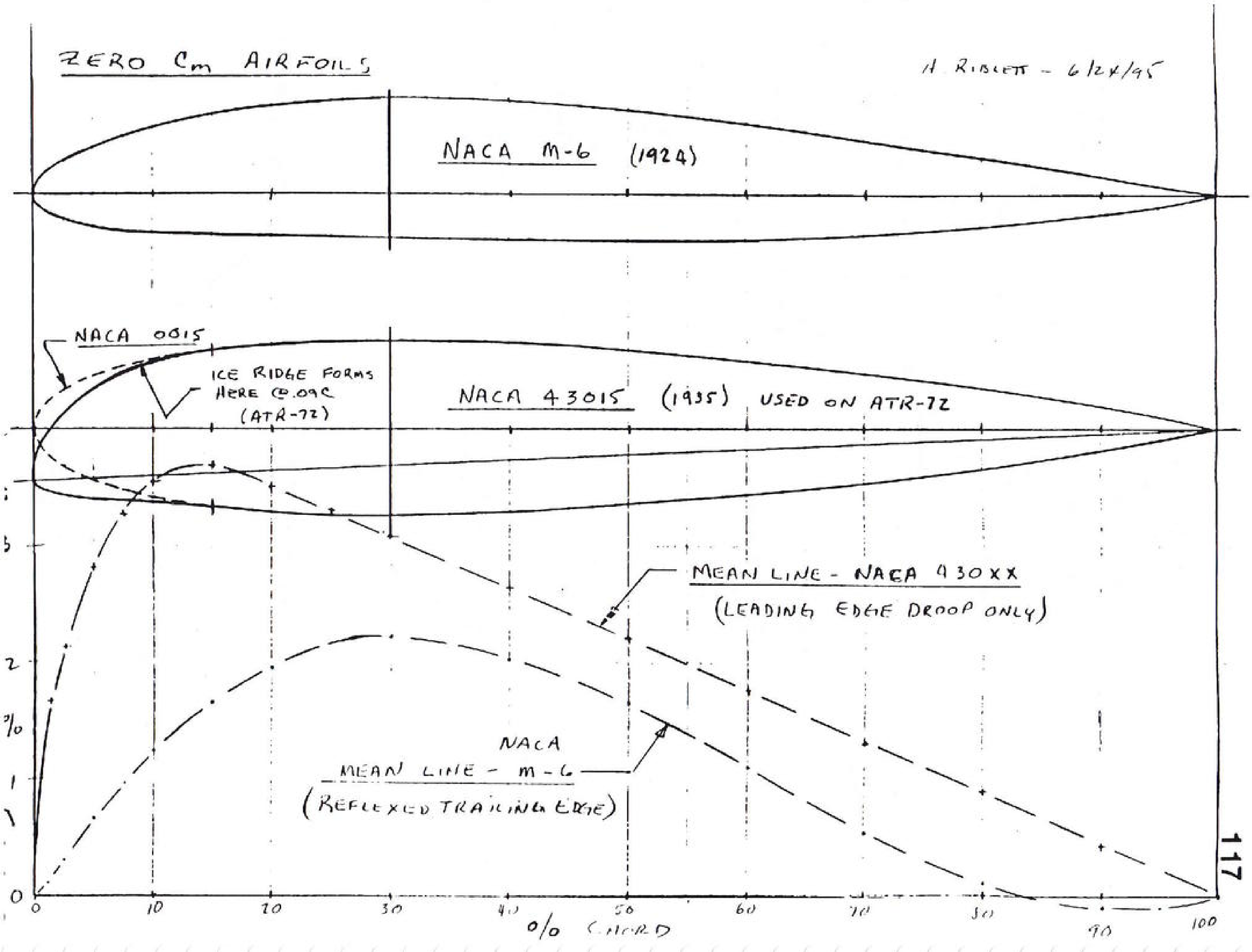
Modified Venture Stall Pattern

TEST RESULTS

Wind tunnel data indicates that the addition of these wing enhancements provides more useable lift than the industry-standard stall strips while drastically softening the stall break. Subsequent flight tests have further demonstrated the departure resistance of the configuration.



Wind Tunnel Results

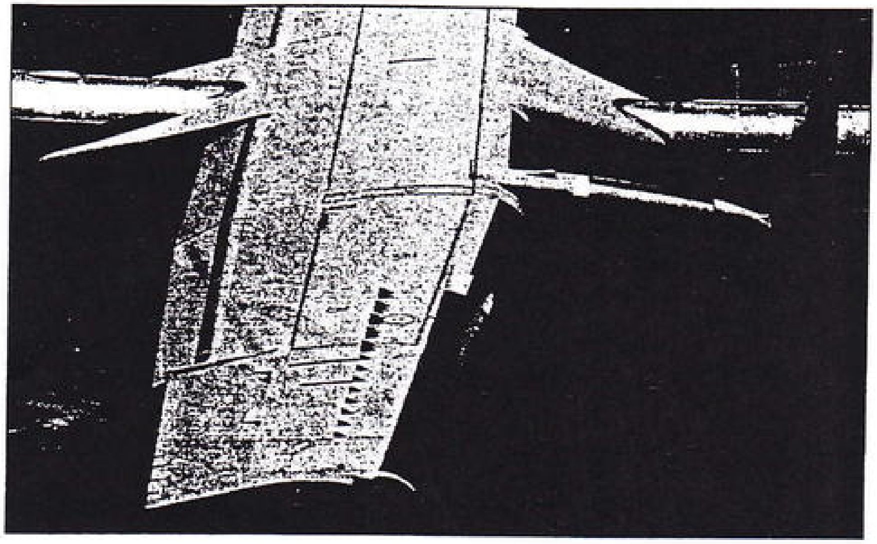

ZERO Cm AIRFOILS
H. RIBLETT - 6/28/95
NACA M-6 (1924)
NACA 0015
ICE RIDGE FORMS HERE @.09C (ATR-72)
NACA 43015 (1935) USED ON ATR-72
MEAN LINE - NACA 430XX (LEADING EDGE DROOP ONLY)
NACA MEAN LINE - M-6 (REFLEXED TRAILING EDGE)
%
% CHORD

U.S. APPROVES NEW ATR DEICER

(ATR72 uses NACA 4301X airfoils)

EDWARD H. PHILLIPS/WASHINGTON

Larger deicer prevents formation of ice forward of the ailerons. Nearly 200 aircraft in North American service are scheduled to be retrofitted by June 1.



The FAA has approved a new deicer for ATR transports and plans to begin investigating icing characteristics of other turboprop regional aircraft later this year.

Certification of the deicer last week occurred following four months of FAA study and evaluation of "all available technical data" by the agency and its French counterpart, the Director Generale de l'Aviation Civile, Daniel Salvano, special assistant at FAA's Aircraft Certification Service, said. These data included flight tests in natural and artificial icing conditions.

The enlarged deicer covers up to 12.5% of the wing's mean aerodynamic chord—nearly twice the area formerly protected against ice accumulation. Flight tests conducted at Edwards AFB, Calif., earlier this month indicate the deicer prevents formation of an ice ridge forward of the ailerons.

A U.A. AIR FORCE KC-135 tanker equipped with water spray equipment deluged the ATR72 in a series of tests that included water droplets up to 200 microns in diameter. The deicers functioned properly with flaps retracted and with flaps extended to 15 deg.—a setting normally used in holding patterns.

A similar ridge of ice is suspected to have caused a loss of lateral control that led to the crash of an American Eagle ATR72-210 near Roselawn, Ind., in October. That aircraft was flying a holding pattern with flaps at 15 deg. when the accident occurred.

During the tests at Edwards AFB in December and again this month, particular attention was paid to ice accumulation with the flaps at 15 deg. At that setting, ice was able to accrete aft of the original deicer. When the flaps were retracted, the airflow above the wing was disrupted, changing aileron hinge moment dramatically. With the new deicer, no adverse effects occurred because the ice ridge failed to form.

ATR officials expect to complete retrofitting new wing deicers on ATR42s and ATR72s by June 1—a deadline established by the FAA in January.

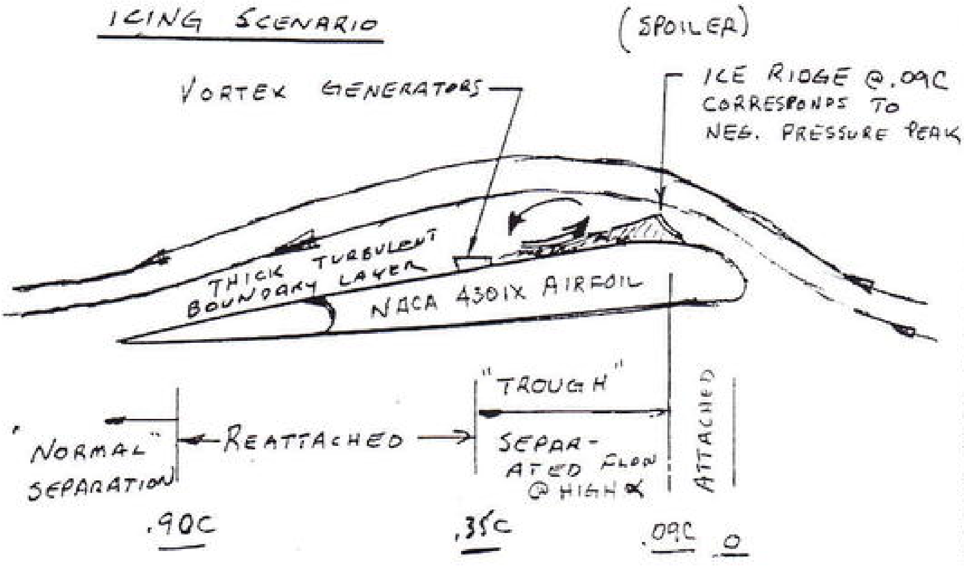
As a result of the ATR accident, FAA officials have notified U.S. and foreign airframe manufacturers that the agency plans this year to test other turboprop-powered regional aircraft. The chief goal will be to determine their susceptibility and reaction to the effects of freezing drizzle and freezing rain, Salvano said. He expects the testing to be completed before the start of icing season.

THE FAA'S TRANSPORT Aircraft Directorate and Small Airplane Directorate will be responsible for selecting test parameters and preparing programs to conduct the experiments. Although in-flight testing similar to that at Edwards AFB will be performed, some tests can be done in wind tunnels, he said. Tailplane icing also will be explored as part of a separate but key program, according to Salvano.

In addition to these restrictions, the FAA is preparing another airworthiness directive intended to impose new restraints upon flight operations with the redesigned deicers, Salvano said. He expects the directive to be issued this month.

Despite the new deicer and pleas from ATR officials, the FAA last week refused to remove restrictions imposed in January affecting flight operations of the ATR42 and ATR72. Pilots must continue to observe the limitations after the deicers are fitted to the aircraft. These include:

- Requiring pilots to monitor cockpit side windows for a unique ice accretion that indicates freezing rain has been encountered.
- Prohibiting use of flaps when holding in icing conditions and requiring immediate disengagement of autopilot in freezing drizzle or rain.
- Leaving flaps deployed if they are extended in freezing drizzle or rain. ■



ICING SCENARIO

ADDENDUM NUMBER 4

An Upper Surface Wing Re-profile for the BD-5

The BD-5 airplanes as originally designed have a well documented, undesirable sharp stall characteristic that has caused numerous accidents, many of them fatal. The main cause of the problem is the use of the infamous NACA 64-212 airfoil section at the wing root. This airfoil is low-cambered and only 12% thick, which largely accounts for its low maximum lift coefficient and also the the sharp loss of lift at the stall, followed by a hysteresis loop during stall recovery. That is, the angle of attack must be decreased considerably in order to obtain flow re-attachment after stall separation has occured, usually with a considerable loss of altitude. Also, the stall is often accompanied by loss of lateral control, resulting in a wild departure. Other factors contribute to the problem, such as the extremely low Reynolds number of this airplane at the stall, less then 1½ million. All in all, the airfoil selection was especially poor for this application, and this is inexcusable.

Short of a complete wing re-design, various methods are available to improve the BD-5 stall performance. For example, BD Micro Technologies offer drooped leading edge nose ribs (and other components) for wings that are still under construction. For completed airplanes, however, several methods of re-profiling the wing shape have been used with good results. Seth Anderson has modified his well-known BD-5 with an upper forward surface re-profile of his own design, extending back to approximately 40% of chord on the top surface, and this has been flying for several years with excellent results. This modification is minimal and relatively simple, and, like other upper surface build-up schemes, stiffens the upper skin to prevent wrinkling. Unfortunately, however, this partial re-profiling essentially reverts the wing profile to a turbulent section, resulting in a slight drag increase. This is only a minor disadvantage, and is of little concern, considering the marked improvement in flight safety that results.

Another scheme for upper surface re-profiling is shown on the attached sketch, dated 11/20/92. In this scheme the complete upper surface is built up from the leading edge to the flap/aileron well lip, at approximately 80% of chord. The flaps and ailerons require no modification and are re-used as is. Also, the existing tip section, NACA 64-218, while improvable, has acceptable stall performance due to its 18% thickness. Thus, a re-profile template is furnished for the wing root only, and the re-profile build-up feathers out to nothing at the wing tip, and at the leading edge and at the flap/aileron well lip. Maximum build-up, at the root, is ½ inch additional thickness.

The "GA" re-profile improves the all-important leading edge profile, adds a small amount of thickness and camber, and moves the loading forward slightly so that the pitching moment coefficient is not affected. The result is a softer stall, higher maximum lift coefficient with and without flaps, and a wider laminar bucket. In addition, the new profile has been designed to maintain constant velocity flow to approximately mid-chord, preserving laminar flow (and low drag). CG limits for the airplane are not affected.

Flight experience with this modification has been excellent. A typical example is the attached report from Bobbie Farmer of Odessa, Texas.

Another option for BD-5 projects under construction is simply to assemble the wings as designed, and then re-profile the top surface before flight.

In the interest of improving flight safety, I offer a full-size template drawing for $10,00 to cover the cost of printing and mailing.

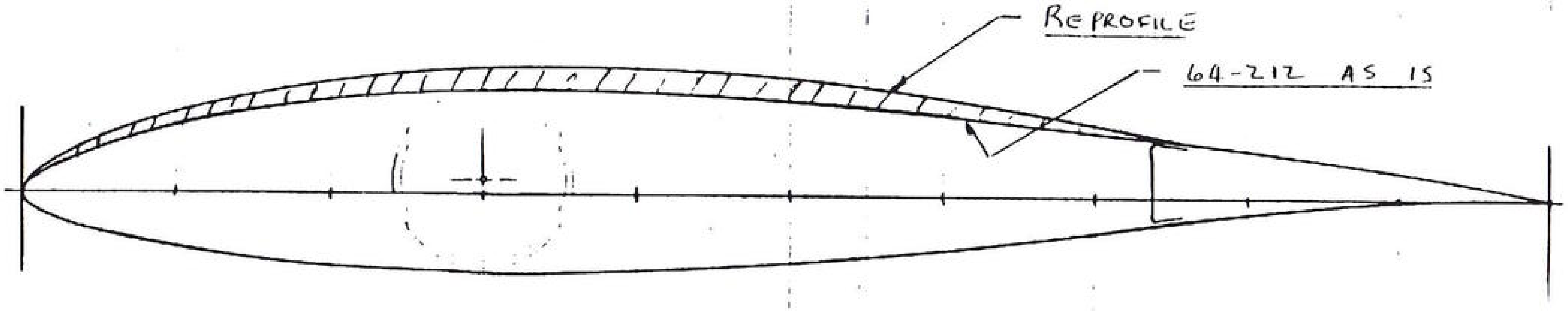
8/16/1995

Harry Riblett, EAA #29576
416 Riblett Lane
Wilmington DE 19808 302/994-0479

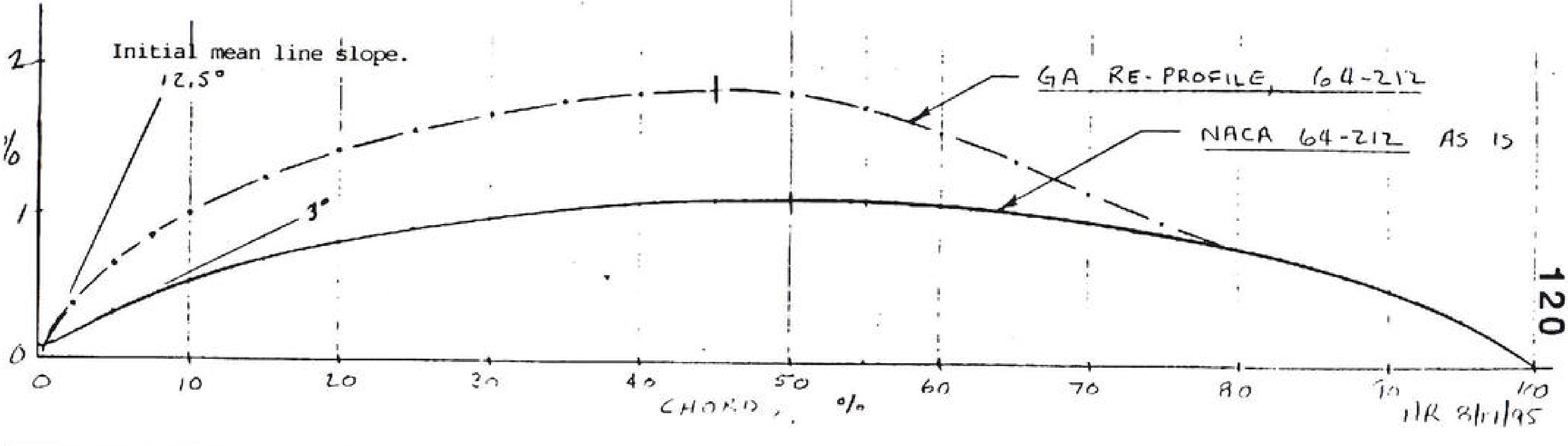
Note. The attached standard table of ordinates can be used to generate the profile for the BD-5 root template (31 inch chord), or for any airplane using the NACA 64-212 airfoil.

GA RE-PROFILE, NACA 64-212 AIRFOIL 13.48% THICK

As TESTED ON BD-5B


REPROFILE
64-212 AS IS

CAMBER PROFILE (MEAN LINE)



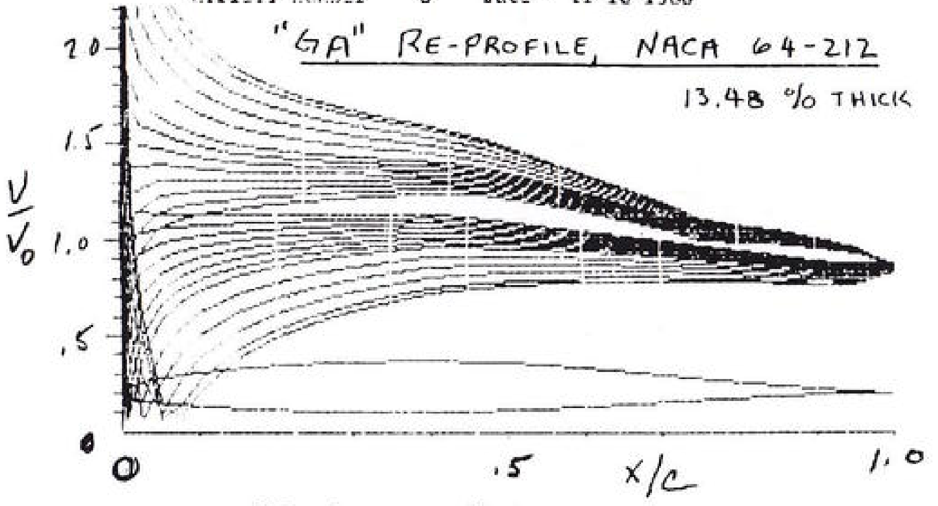
"GA" RE-PROFILE, NACA 64-212

Dec 88 Sta	NACA 64-212 Mod Ax100 inch upper	Sta	lower
0.000	0.000	0.000	0.000
0.238	0.836	0.200	-0.532
0.713	1.426	0.582	-0.925
1.188	1.835	0.841	-1.105
2.375	2.588	1.353	-1.379
4.750	3.705	2.618	-1.846
7.125	4.540	5.132	-2.491
9.500	5.214	7.636	-2.967
14.250	6.270	10.135	-3.352
19.000	7.076	15.128	-3.945
23.750	7.686	20.114	-4.376
28.500	8.128	25.097	-4.680
33.250	8.413	30.079	-4.871
38.000	8.540	35.059	-4.948
42.750	8.481	40.039	-4.910
47.500	8.218	45.018	-4.703
52.250	7.741	50.000	-4.377
57.000	7.093	54.984	-3.961
61.750	6.314	59.971	-3.477
66.500	5.432	64.961	-2.944
71.250	4.478	69.955	-2.378
75.047	3.700	74.953	-1.800
80.045	2.825	79.955	-1.233
85.038	2.054	84.962	-0.708
90.027	1.303	89.973	-0.269
95.013	0.604	94.987	+0.028
100.000	0.000	100.000	0.000

Airfoil number 0 Date - 11-18-1988

"GA" RE-PROFILE, NACA 64-212

13.48 % THICK

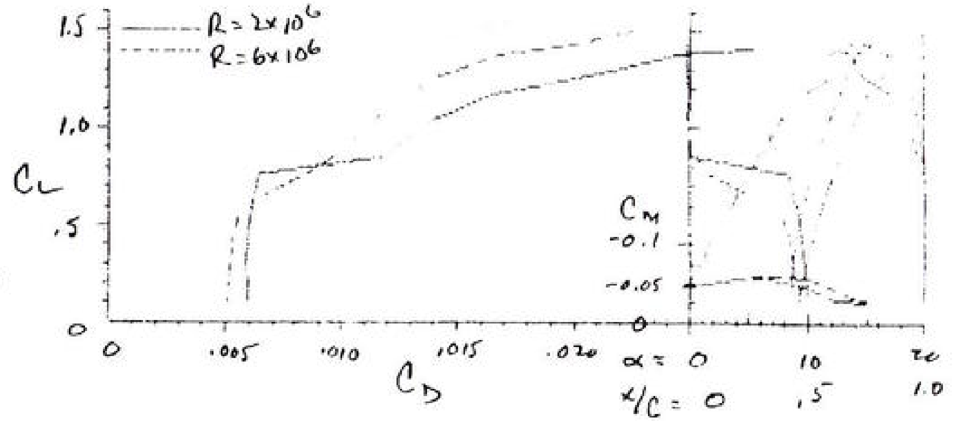


VELOCITY DIAGRAM

Airfoil number 0 Date - 11-18-1988

GA RE-PROFILE, NACA 64-212
(EPPLER PROGRAM)

13.48 %
THICK



R. B. (Bobby) Farmer
P.O. Box 12176
Odessa Texas 79768   915/563-2732

(Retyped)

8/11/95

Harry Riblett
416 Riblett Lane
Wilmington DE 19808

Dear Harry:

Like I told you, I would let you know how the (top surface) scab-on wing airfoil rework came out (on my BD-5B).

It lowered my stall speed from 80 MPH to 67MPH. The stall is now something like an Ercoupe, kind of "mushing" like. Top end, the same- no loss there. The warm day take-off distance was 2500 ft, and is now about 1600 ft. I still hold it down until I get about 85 MPH, and then lift off. However, I now notice that the acceleration is quicker from 60 to 85 MPH than it used to be, probably because the wing is now generating more lift on the take-off roll than before, thus the take-off roll is shorter. Time to rotation is now only 16 seconds @ 1600 ft, with OAT @ 86 degrees, density altitude 6200 ft. Our airport is at 3000 ft.ASL.

For the rework, I left the wings on the ship, and cleaned the top surface of the wing of all paint, to the bare aluminum, and scrubbed this with APX. I used 2 lb. density foam, 1/2 in thick, covered with two layers of glass cloth. I put on the first layer lengthwise, and the second layer 90 degrees to the first, and finished off with two coats of polyester. I then reinforced the leading edge with two layers of fine weave 3-inch glass tape, staggered, which extended back about 2 inches from the leading edge on the lower surface. The total cost of all materials was $105.00 from Wicks Aircraft, and it took me 44 hours to put it all together. The airplane empty weight was 467 lbs, and is now 471 lbs, so the rework added only 4 lbs to the empty weight.

My ship is a BD-5BDH, S/N 19, N-6782F, which I built myself. It has the standard (non-stretched) body length. The wing span is 21½ ft. It has a non-turbo Honda engine with a 46x66 Hegy prop. It has Matco wheels and disc brakes.

Thanks again, Harry-   Bob Farmer

P.S. I also wrote to Rich Perkins and gave him all the info and pictures for the BD-5 network newsletter.

ADDENDUM NUMBER 5

A Critique of the NASA NLF(1)-0115 Airfoil

The July- August 1995 issue of the "Journal of Aircraft" (AIAA) contains an article by Selig, Maughmer, and Somers, that presents a "new NASA" airfoil, NLF(1)-0115, a "Natural Laminar Flow Airfoil for General Aviation Applications". As it turns out, the airfoil was designed by Herr Doktor Richard Eppler, and I have corresponded with him about it. The airfoil and its mean line (camber profile) are shown on figure 1. After plotting these, I saw nothing frightening about the thickness distribution, although the point of maximum thickness is relatively far aft at .42C, but I was concerned about three things in the mean line:

1. The nose of the airfoil contains excessive leading edge droop, somewhat like the NACA 23015 airfoil, which has a sharp stall due to excessive leading edge droop. Therefore, I suspected that NLF(1)-0115 may have a similar sharp stall.
2. A dip appears in the camber profile at .60C, similar to the NLF(1)-0215F airfoil, also due to Eppler. As we have seen previously, this area of negative lift extends the laminar run slightly (hence the moniker "natural" laminar flow), however the negative lift produced by the camber dip results in a loss of efficiency, a point that I suspect Dr. Eppler has not considered.
3. Pronounced aft camber appears in the airfoil from .65C rearward, and I suspected a relatively high pitching moment coefficient (Cm) as a result. This also would probably cause high aileron hinge moments (stiff ailerons), although this would probably be hard to prove without flight tests.

A computer performance analysis confirmed my fears about this airfoil, especially the sharp stall at 13 degrees angle of attack. Accordingly, in an attempt to track down the cause of the poor performance, I designed a new airfoil for comparison, GA 42E-215, figure 2. This new airfoil looks like GA 40A215 on figure 1, except the point of maximum thickness is at .42C rather than .40C. The new airfoil uses the unoffensive Eppler thickness distribution from NLF(1)-0115, and combines it with the proven GA-2 mean line from "GA Airfoils". This should give us a direct comparison of the efficacy of the Eppler mean line versus the GA-2 mean line.

Figure 3 shows that the performance of GA 42E-215 is indeed superior to the performance of NASA NLF(1)-0115, including a soft stall rather than a sharp stall, a wider laminar bucket, and a 30-40% reduction of pitching moment coefficient. The profile drag , as expected, is the same, since both airfoils use the same thickness distribution. The obvious conclusion is thac the mean line of NASA NLF(1)-0115 is poorly designed.

If an even wider laminar bucket is desired for operational flexibility, GA 40A215 may be used, but the trade-off is a slightly higher profile drag coefficient. See figure 3.

In spite of the indicated superiority of GA 42E-215, Dr. Eppler does not like it, because it was not designed by the modern "inverse" method for designing airfoils.

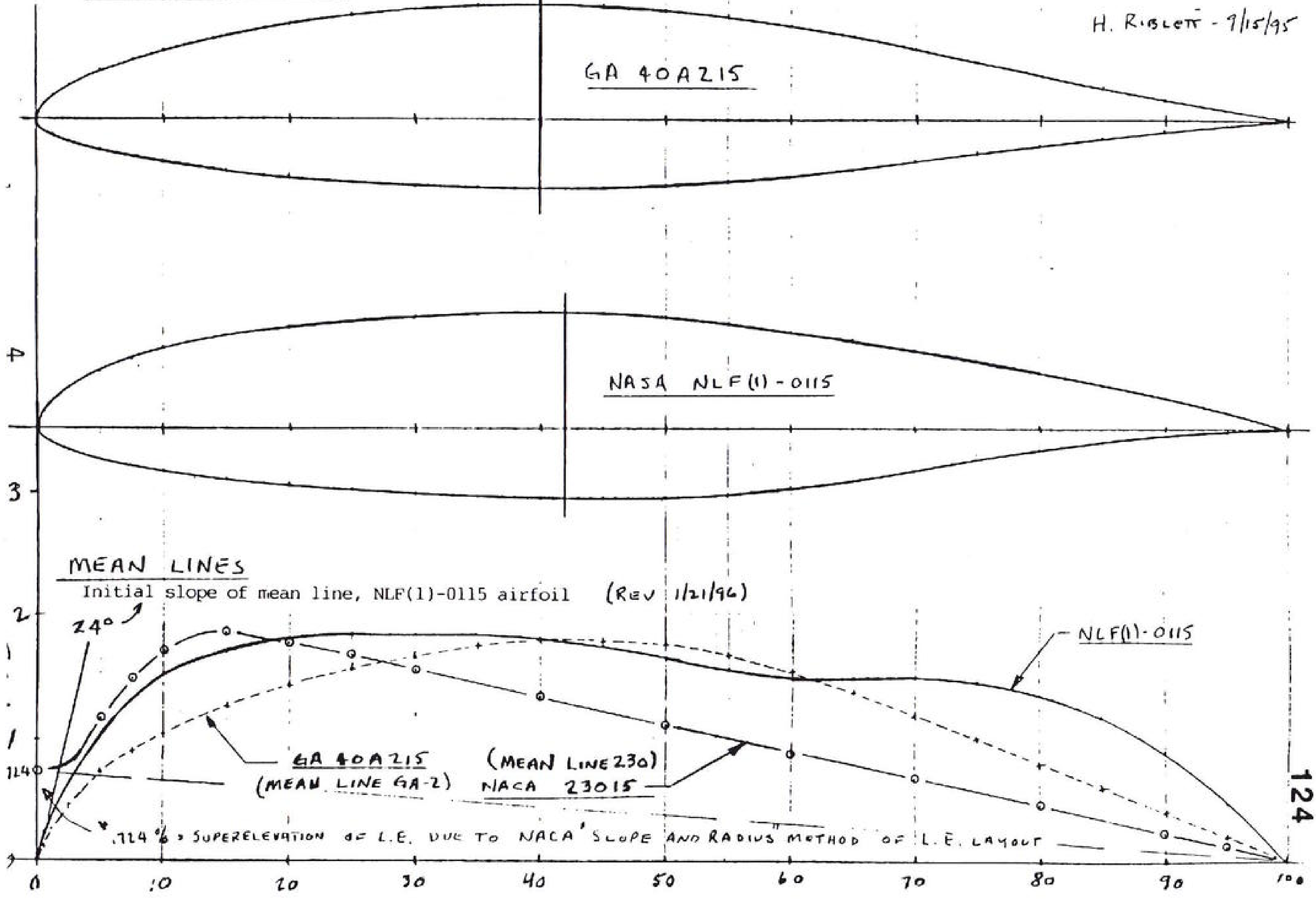
You be the judge.

Harry Riblett
416 Riblett Lane
Wilmington DE 19808

2/14/1996

AIRFOIL COMPARISON

H. RIBLETT - 9/15/95

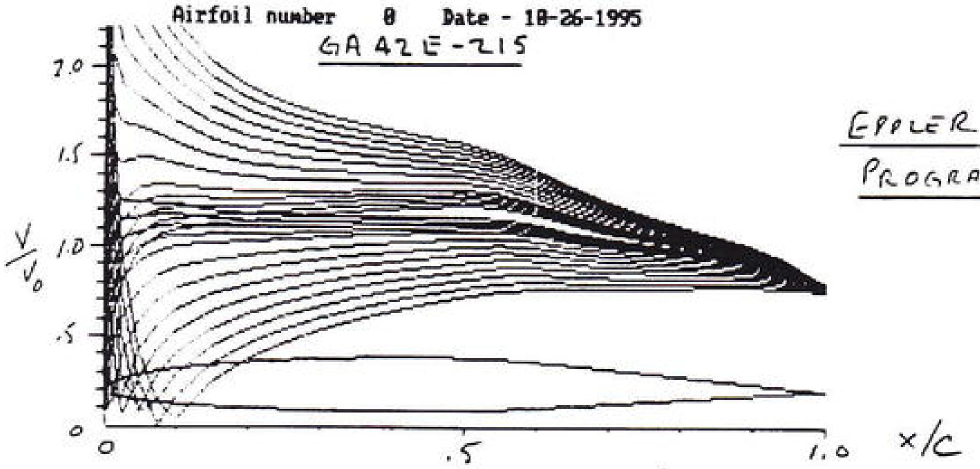

GA 40A215
NASA NLF(1)-0115
4
3
2
1
.1240
24°
NLF(1)-0115
GA 40A215
(MEAN LINE GA-2)
(MEAN LINE 230)
NACA 23015
.124% = SUPERELEVATION OF L.E. DUE TO NACA "SLOPE AND RADIUS" METHOD OF L.E. LAYOUT
0
10
20
30
40
50
60
70
80
90
100

MEAN LINES

Initial slope of mean line, NLF(1)-0115 airfoil (REV 1/21/96)

EPPLER

Oct 95	42E-015	Camber	42E-212		42E-215		42E-218	
Sta	Ord	GA-2	upper	lower	upper	lower	upper	lower
0.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
0.25	0.805	0.060	0.704	-0.584	0.865	-0.745	1.026	-0.906
0.50	1.100	0.116	0.996	-0.764	1.216	-0.984	1.436	-1.204
0.75	1.380	0.169	1.273	-0.935	1.549	-1.211	1.825	-1.487
1.25	1.800	0.263	1.703	-1.177	2.063	-1.537	2.423	-1.897
2.50	2.500	0.450	2.450	-1.550	2.950	-2.050	3.450	-2.550
5.00	3.530	0.715	3.539	-2.109	4.245	-2.815	4.951	-3.521
7.50	4.290	0.892	4.324	-2.540	5.182	-3.398	6.040	-4.256
10.00	4.870	1.023	4.919	-2.873	5.893	-3.847	6.867	-4.821
15.00	5.705	1.244	5.808	-3.320	6.949	-4.461	8.090	-5.602
20.00	6.310	1.420	6.468	-3.628	7.730	-4.890	8.992	-6.152
25.00	6.780	1.557	6.981	-3.867	8.337	-5.223	9.693	-6.579
30.00	7.130	1.663	7.367	-4.041	8.793	-5.467	10.219	-6.893
35.00	7.370	1.737	7.633	-4.159	9.107	-5.633	10.581	-7.107
40.00	7.485	1.780	7.768	-4.208	9.265	-5.705	10.762	-7.202
45.00	7.465	1.792	7.764	-4.180	9.257	-5.673	10.750	-7.166
50.00	7.300	1.764	7.604	-4.076	9.064	-5.536	10.524	-6.996
55.00	6.940	1.672	7.224	-3.880	8.612	-5.268	10.000	-6.656
60.00	6.355	1.537	6.621	-3.547	7.892	-4.818	9.163	-6.089
65.00	5.595	1.374	5.850	-3.102	6.969	-4.221	8.088	-5.340
70.00	4.770	1.189	5.005	-2.627	5.959	-3.581	6.913	-4.535
75.00	3.940	0.991	4.143	-2.161	4.931	-2.949	5.719	-3.737
80.00	3.110	0.793	3.281	-1.695	3.903	-2.317	4.525	-2.939
85.00	2.295	0.595	2.431	-1.241	2.890	-1.700	3.349	-2.159
90.00	1.495	0.396	1.592	-0.800	1.891	-1.099	2.190	-1.398
95.00	0.690	0.198	0.750	-0.354	0.888	-0.492	1.026	-0.630
100.00	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000



EPPLER PROGRAM

GA 42E-215

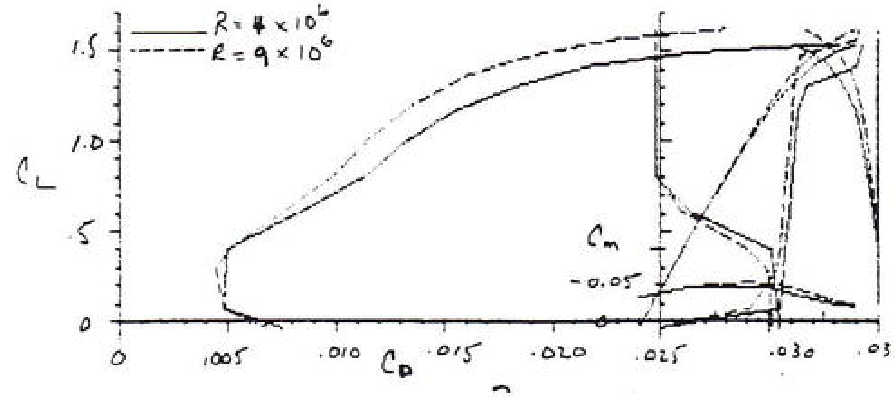
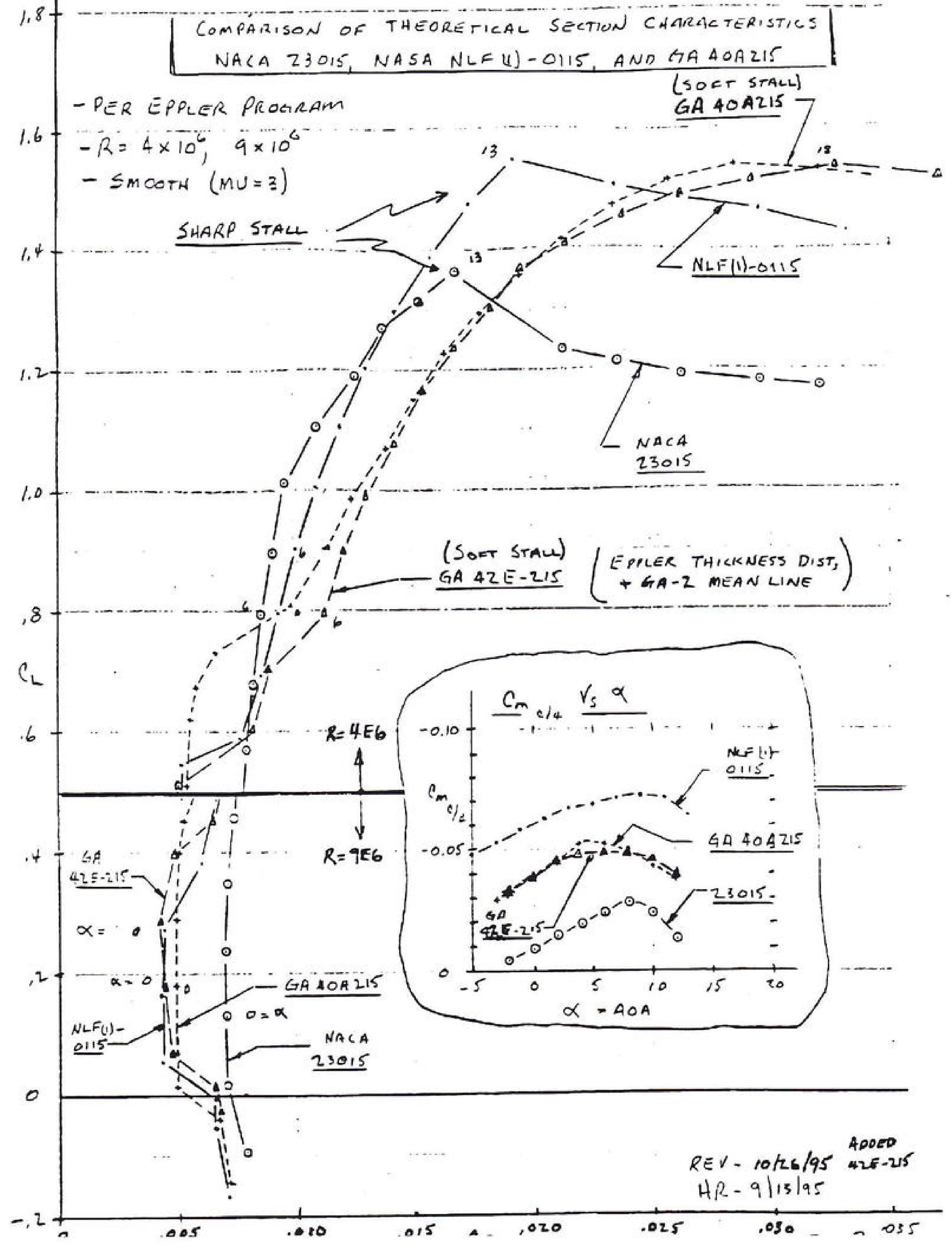


FIG 2



REV - 10/26/95 ADDED 42E-215
HR - 9/15/95

ADDENDUM NUMBER 6

Harry Riblett
416 Riblett Lane
Wilmington DE 19808
302/994-0479

2/8/1996

National Transportation Safety Board
490 L'Enfant Plaza East
Washington DC 20594

Attn: Mr. John Clark, Chief, Vehicle Performance Division

Subject: American Eagle ATR-72 crash at Roselawn Indiana 10/31/1994

Dear Mr. Clark:

My letter to you 9/7/95 explained why the ice ridge on the ATR-72 wing, which was the immediate cause of loss of control, formed at exactly the 9% chord position. In you response letter 1/25/1996 you ignored this point completely, which is unfortunate, for that is the key to the puzzle. Therefore, I am enclosing additional data showing that the cause of the ice ridge, which caused the accident, is due to an anomaly of the NACA 430xx airfoil, that is, a pronounced negative pressure peak at the 9% chord position, exactly where the ice ridge forms. More conventional airfoils do not have this problem. Therefore, the accident was caused by the poorly designed, unusual, and fortunately rarely used, NACA 430xx airfoils used on this airplane.

This ill-conceived airfoil design was part of a test program conducted by NACA Langley on "airfoils having the camber unusually far forward", merely to see what effects the extreme forward loading would have on airfoil performance- hopefully, low pitching moment coefficients. Figure 2 shows the forward loading schedule, and compares it to a more conventional, uniformly loaded airfoil, the NACA 64-415, used on the Fokker F-27 "Friendship". The theortical pressure profiles predicted by NACA for these airfoils, figure 4, shows a pronounced peak at .09C for the 430xx airfoil, and no peak whatever for the 64-415 airfoil. The actual pressure distributions (velocity ratio curves) for these two airfoils obtained by computer performance analysis, figure 6, confirms the predicted profiles.

The results of the 1935 NACA wind tunnel tests, figure 5, show that the airfoil moment coefficient (unrelated to the Roselawn accident) is indeed low, as anticipated. However, the sharp and dangerous stall on these airfoils, as bad as any airfoil ever tested by NACA, is unacceptable, therefore the tests must be considered a failure. Unfortunately, these test airfoils came to be used on actual airplanes, with generally disappointing results, especially on prop-driven twins.

The ice ridge forms at the negative pressure peak (.09C) because that is where the adiabatic cooling is the greatest. This is analagous to ice formation in the throat of a carburetor. In fact, the top surface of an airplane wing is simply a one-sided venturi. From that point onwards, further conjecture as to the exact effect the ice ridge had on wing performance, aileron hinge moments, etc, are moot. The point is, the unusual pressure peak on the 430xx airfoils at .09C caused the ice ridge to form at that point, and that downed the airplane. Therefore, the NACA 430xx airfoil is at fault.

Accordingly, there may be some merit for the FAA to ban airplanes using the NACA 430xx airfoils from flying in freezing rain and drizzle (figure 1), but not all turboprop regional airliners should be so banned. To issue a blanket prohibition is heavy-handed, imperialistic, and ignores the facts in the case.

For confirmation of this airfoil data, check with someone other than:
-NASA, who is responsible for the idiotic 430xx airfoil;
-ATR (AIR), who made the mistake of choosing this airfoil for their airplanes;
-FAA, who made the mistake of certifying the ATR airplane with this airfoil on it.

By copy of this letter, I request FAA's Mr. Salvano to include this letter as a public comment in the docket regarding this accident, and also to include it in the FAA's icing symposium scheduled for May 6-8 at Springfield, VA.

In the interest of improving flight safety,

Harry Riblett Harry Riblett

CC: w/encl.

Mr. Dan Salvano, FAA, Washington DC 20591

Mr. Tony Broderick, FAA, Washington DC 20591

Mr. Andrew Cebula, V.P., Gov't &Industry Affairs, National Air Transportation Assn.

Mr. Walt Coleman, President, Regional Airline Assn.

Mr. Edward Phillips, AW&ST, Washington DC 20005

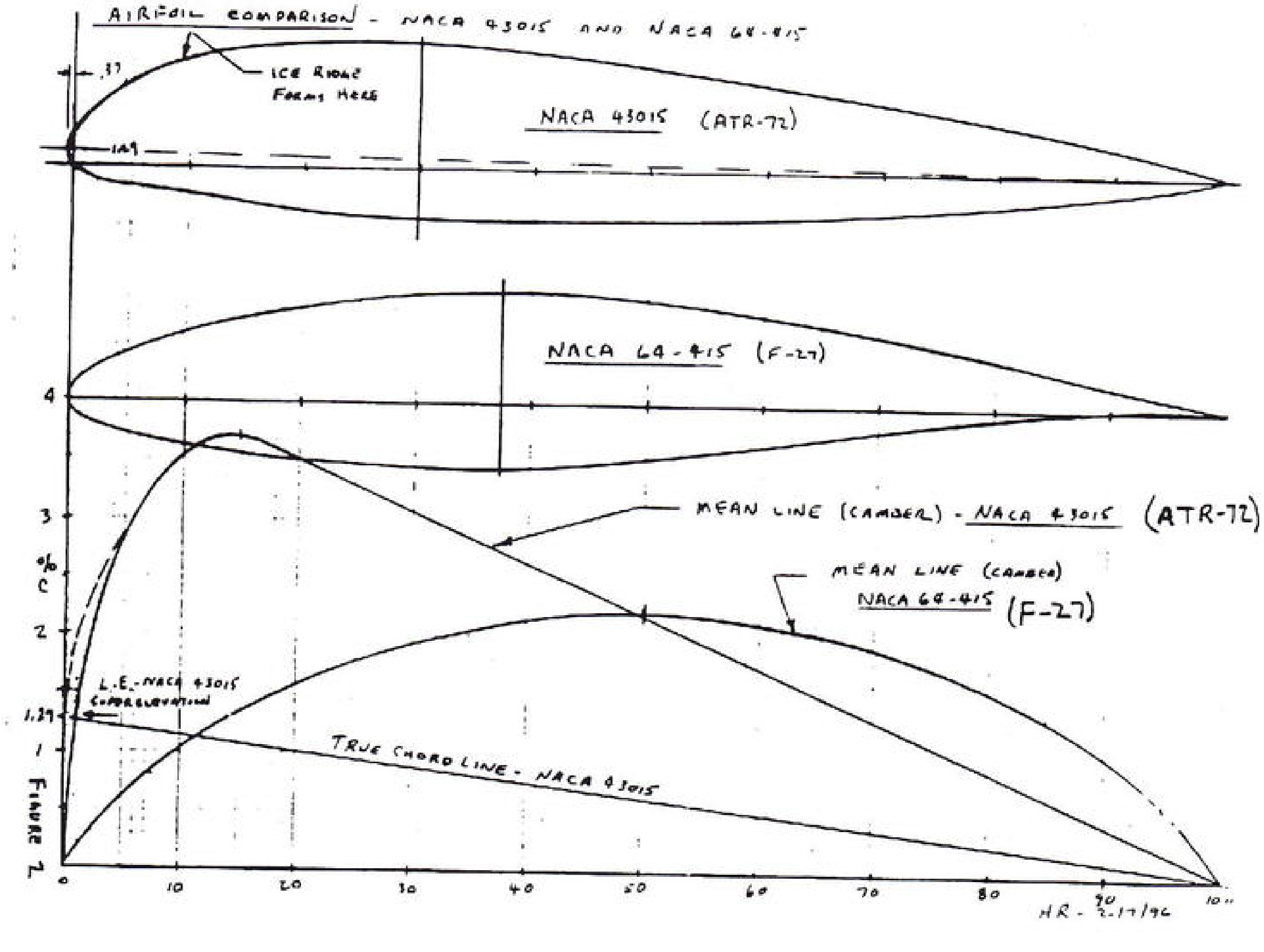

AIRFOIL COMPARISON - NACA 43015 AND NACA 64-415
.37
ICE RIDGE FORMS HERE
NACA 43015 (ATR-72)
1.49
NACA 64-415 (F-27)
4
3
% C
2
1.39
1
MEAN LINE (CAMBER) - NACA 43015 (ATR-72)
MEAN LINE (CAMBER) NACA 64-415 (F-27)
L.E.-NACA 43015 CORRELATION
TRUE CHORD LINE - NACA 43015
0
10
20
30
40
50
60
70
80
90
100

FIGURE 2

HR - 2-17/96

FAA TO BAN TURBOPROPS IN FREEZING RAIN, DRIZZLE

EDWARD H. PHILLIPS/WASHINGTON

Citing an unsafe condition not covered by current certification rules, the FAA plans to issue an airworthiness directive by May forbidding virtually all turboprop-powered regional airline aircraft from flying in freezing rain and drizzle.

The directive will prohibit at least 18 different aircraft, including all turboprop regional transports in U.S. service, from operating in freezing rain and drizzle by amending each aircraft's approved Airplane Flight Manual, said Dan P. Salvano, manager of the FAA's Rotorcraft Directorate, Ft. Worth. The FAA has set Mar. 7 as the deadline for receiving comments, and "is very reluctant" to extend the comment period beyond that date, he said.

Aircraft chiefly affected by the directive include the Jetstream 31/41, ATR42/72, de Havilland DHC-7/DHC-8, Dornier 228/328, Embraer EMB-120, Saab 340/2000 series, Fairchild SA226/227 and Beechcraft Model 99 series, B200 and 1900 series. These aircraft meet icing certification criteria under Appendix C of Part 25 of the Federal Aviation Regulations.

The directive would require changes to the Airplane Flight Manual that would:

- Prohibit use of the autopilot when ice has formed aft of the wing deicers, or when an unusual lateral trim condition exists.
- Require that all ice lights be operative prior to flight into icing conditions at night, irrespective of exceptions contained in a particular aircraft's Master Minimum Equipment List.
- Limit use of flaps and forbid use of the autopilot if ice is observed forming aft of the wing deicers.

In addition, the directive mandates that pilots use approved procedures to immediately exit freezing drizzle or rain conditions and that they make every effort to avoid inadvertently flying into such regions.

Aircraft such as the Saab 340 are not certificated to fly in freezing rain or drizzle.

The FAA's latest action stems from the crash of an American Eagle ATR72-200 at Roselawn, Ind. (*AW&ST* Nov. 14, 1994, p. 28; Nov. 7, 1994, p. 36). Although the U.S. National Transportation Safety Board has not determined the probable cause of the accident, analysis indicates that an ice ridge formed aft of the wing deicers, causing a rapid roll and subsequent loss of control by the flight crew (*AW&ST* Jan. 2, 1995, p. 28).

THAT ACCIDENT has not only reinforced concern about the danger of freezing rain and drizzle, but has forced the FAA to consider expanding Appendix C of Part 25 beyond its current breadth. Part 25 governs certification of Transport Category aircraft.

Existing regulations require manufacturers to demonstrate that an aircraft can operate safely in water droplets having an average diameter of 20-50 microns. The diameter of droplets encountered by the American Eagle ATR72 probably exceeded 200 microns, according to the NTSB.

To learn more about freezing drizzle and rain, the FAA is planning an international icing symposium currently scheduled for May 6-8 in Springfield, Va. Key topics to be discussed include how freezing rain and drizzle should be defined, problems associated with forecasting its presence and development of advanced ice detection systems capable of providing an early warning to pilots. The use of outside visual cues also will be discussed, Salvano said.

"We are not expecting to make any declarations about freezing rain and drizzle after the meeting concludes," but participants will be encouraged to openly discuss the issue from an operational standpoint, according to Salvano. He expects the Russian delegation to provide important data on freezing precipitation because of their extensive flying experience in severe weather conditions.

The FAA's directive essentially will make all turboprop regional aircraft "equal" by prohibiting flight in freezing rain and drizzle, said Alain Brodin, president of marketing for Aero International Regional (AIR) Marketing Inc. AIR is a sales consortium of the former ATR, Jetstream and Avro companies.

Brodin said the entire U.S. fleet of more than 158 ATR42 and ATR72 aircraft have been fitted with larger wing deicers as well as ice evidence probes mounted outside of the captain's window. The probe is designed to accrete ice and serve as a visual cue that the aircraft has encountered icing conditions for which it is not certificated, Brodin said.

The FAA is still conducting special ground-based tests designed to determine which regional transports are susceptible to control problems under severe icing conditions. In-flight testing using a specially modified U.S. Air Force NKC-135A tanker aircraft is available for manufacturers, but the FAA is not mandating its use, Salvano said.

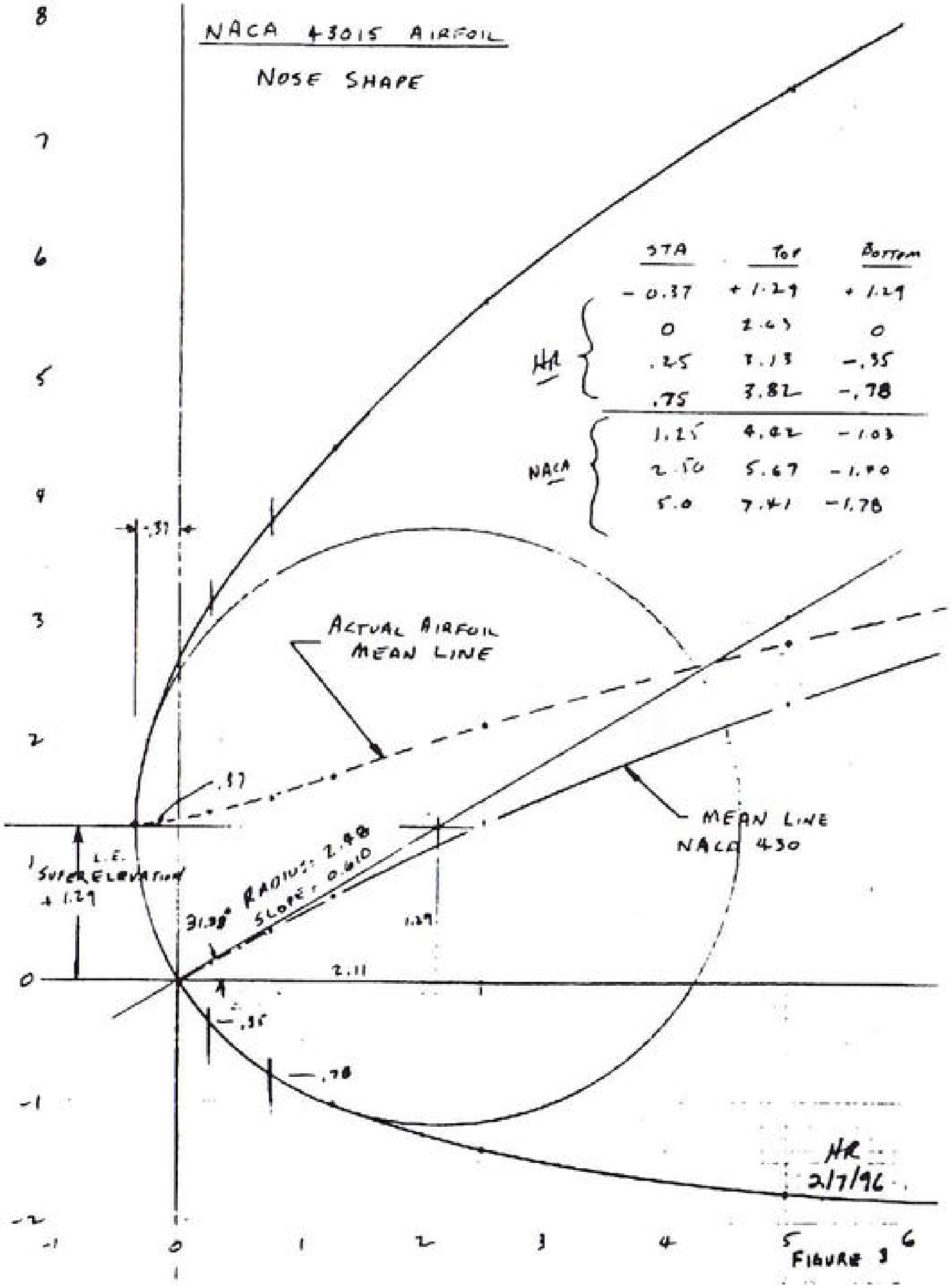
OPERATION OF THE TANKER is funded through Fiscal 1996, "but we have no guarantees beyond this year" that the unique aircraft will remain operational, Salvano said. It costs manufacturers about $250,000 to $500,000 to charter the tanker for icing tests.

In addition to the ATR72-200, an Embraer EMB-120 Brasilia is the only other regional aircraft that has flown behind the NKC-135A. The in-flight tests provide a "critical data point" for manufacturers and the FAA that cannot be derived from ground-based testing, Salvano said. ■

Figure 1

NACA 43015 AIRFOIL

NOSE SHAPE


8
7
6
5
4
3
2
1
0
-1
-2
-1
0
1
2
3
4
5
6
.37
.37
ACTUAL AIRFOIL MEAN LINE
MEAN LINE NACA 430
L.E. SUPERELEVATION + 1.29
31.38° RADIUS: 2.48
SLOPE: 0.610
1.29
2.11
.35
.78

	STA	TOP	BOTTOM
HR	- 0.37	+ 1.29	+ 1.29
	0	2.63	0
	.25	3.13	-.35
	.75	3.82	-.78
NACA	1.25	4.02	-1.03
	2.50	5.67	-1.40
	5.0	7.41	-1.78

HR
2/7/96

FIGURE 3

"COMPREHENSIVE REFERENCE GUIDE TO AIRFOIL SECTIONS FOR LIGHT AIRCRAFT" - EAA BOOKSTORE #21-37111 $19.95

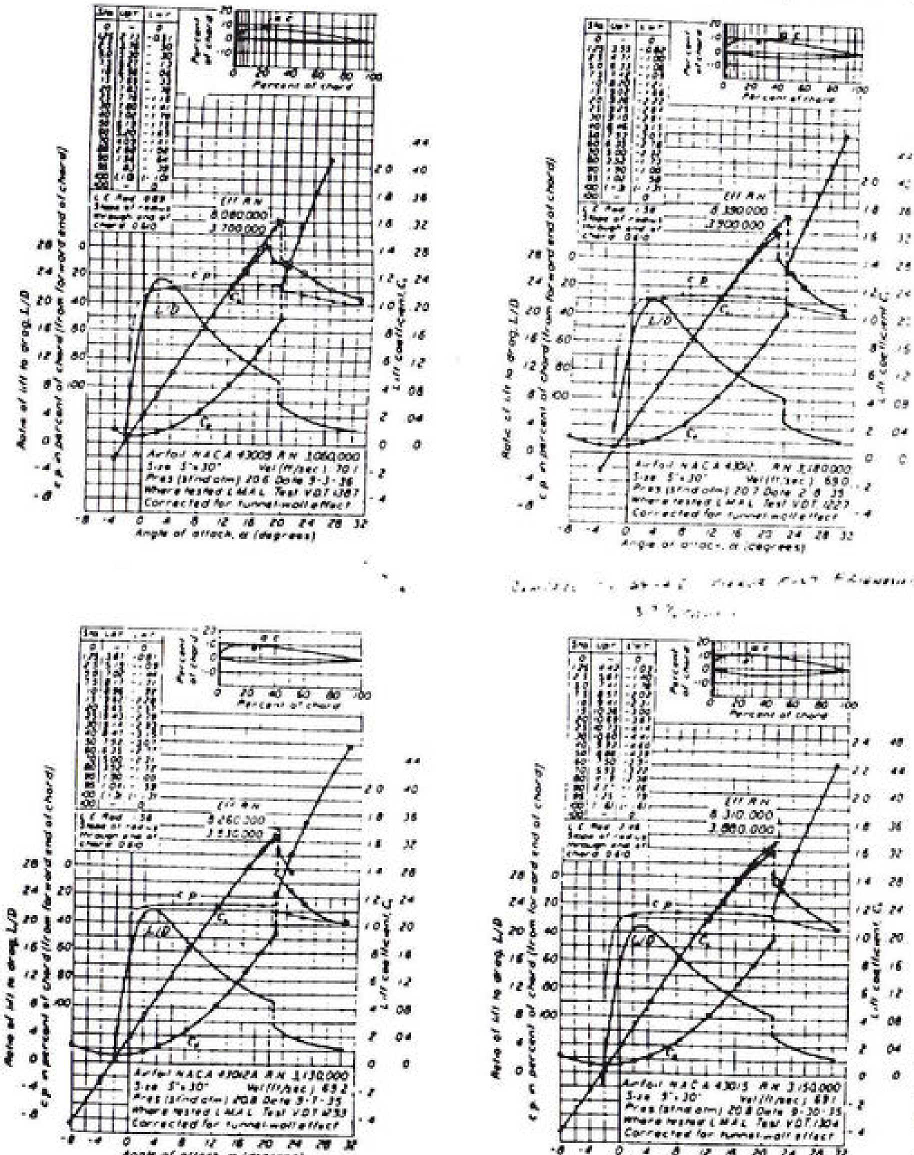
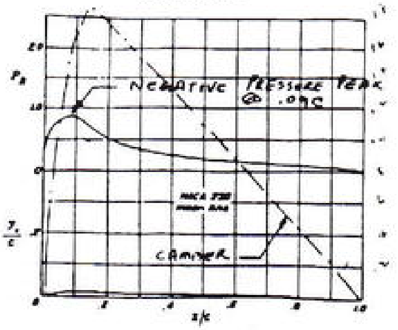


FIGURE 5



c_{l_i} = 0.30	α_i = 1.05°	$c_{m_{c/4}}$ = − 0.014		
x (per cent c)	y_c (per cent c)	dy_c/dx	P_R	$\Delta v/V = P_R/4$
0	0	0.30508	∞	0
1.25	0.357	0.26594	0.528	0.132
2.5	0.666	0.22929	0.673	0.168
5.0	1.155	0.16347	0.791	0.198
7.5	1.492	0.10762	0.853	0.213
10	1.701	0.06174	0.859	0.215
15	1.838	− 0.00009	0.678	0.170
20	1.767	− 0.02203	0.519	0.130
25	1.656	− 0.02208	0.419	0.105
30	1.546		0.351	0.090
40	1.325		0.274	0.069
50	1.104		0.217	0.054
60	0.883		0.177	0.044
70	0.662		0.144	0.036
80	0.442		0.105	0.026
90	0.221		0.069	0.017
95	0.110		0.042	0.011
100	0		0	0

Data for NACA Mean Line 230

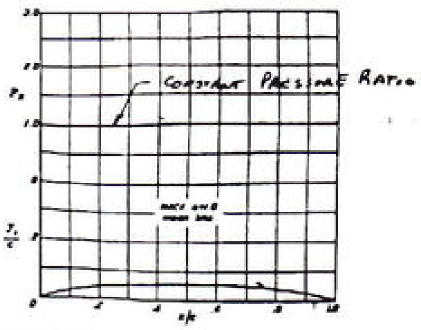
DOUBLE FOR MEAN LINE 430 - (ATR-72)

"THEORY OF WING SECTIONS"

ABBOTT + VON DOENHOFF

FIGURE 4

MULTIPLY BY (0.4) FOR -4XX CAMBER



c_{l_i} = 1.0	α_i = 0°	$c_{m_{c/4}}$ = − 0.250		
x (per cent c)	y_c (per cent c)	dy_c/dx	P_R	$\Delta v/V = P_R/4$
0.25	0.115		4.3°	
0.5	0.250	0.42120		
0.75	0.350	0.38875		
1.25	0.535	0.34770		
2.5	0.930	0.29155		
5.0	1.580	0.23430		
7.5	2.120	0.19995		
10	2.585	0.17485		
15	3.365	0.13805		
20	3.980	0.11030		
25	4.475	0.08745		
30	4.860	0.06745		
35	5.150	0.04925	1.000	0.250
40	5.355	0.03225		
45	5.475	0.01595		
50	5.515	0		
55	5.475	− 0.01595		
60	5.355	− 0.03225		
65	5.150	− 0.04925		
70	4.860	− 0.06745		
75	4.475	− 0.08745		
80	3.980	− 0.11030		
85	3.365	− 0.13805		
90	2.585	− 0.17485		
95	1.580	− 0.23430		
100	0			

Data for NACA Mean Line a = 1.0

(F-27) -2XX -4XX -6XX

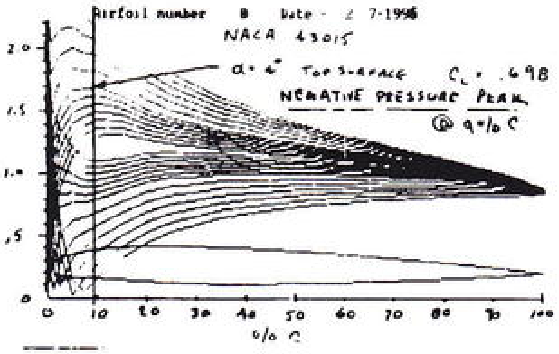
-2XX = 1.1°
-4XX = 2.2°
-6XX = 3.3°
8XX = 5°

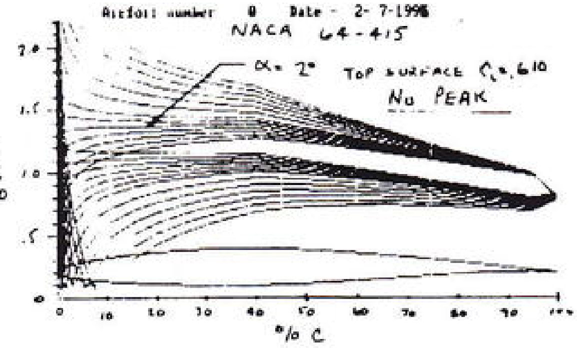
Reduced by Slope + Radius

4.802°
9.537°
14.145°

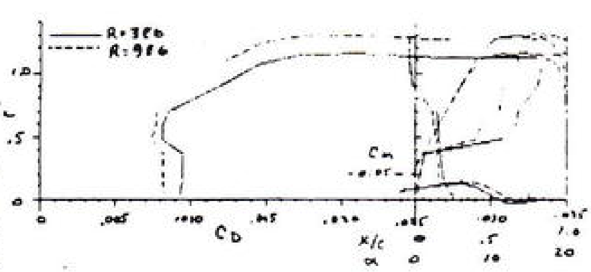
PERFORMANCE COMPARISON - NACA 43015 AND NACA 64-415 AIRFOILS (PER "EPPLER" PROGRAM)

VELOCITY PLOTS





BOUNDARY LAYER SUMMARY



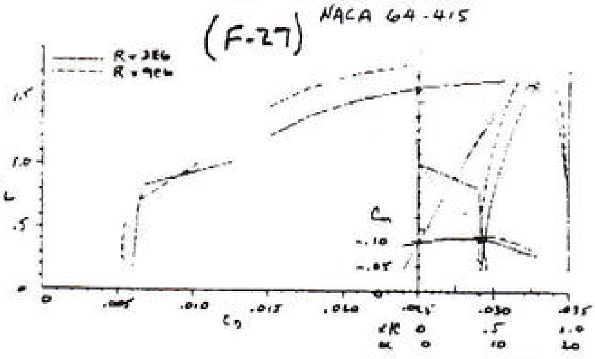


FIGURE 6

About the Author

Harry Riblett, 64 , EAA#29576, is a Mechanical Engineer recently retired from Hercules Inc. in Wilmington, Delaware. He holds a Commercial ASEL pilot's license, and has been active in sport aviation and EAA activities for many years, He is an ex-president of Wilmington Chapter #240 and is currently a chapter Technical Advisor. He has built two homebuilt airplanes, a Starduster II and a CUBy, flies both regularly, and is working on a (nearly finished) single seat original design, as yet unnamed. The new project is a high-wing 350lb. pusher, tricycle gear, with folding wings, and is of typical "Cub" construction.

Harry got his early interest in aviation from his father, who operated a G.I. flight school at Wilmington airport immediatly following WW-II. Harry served as a Reservist Radar Officer/ Air Controller on a USN cruiser during the Korean War. He subsequently worked a dozen or so years with civilian companies manufacturing, testing, and installing aircraft runway arresting gears.

His interest in airfoil design prompted him to visit NASA Langley in 1986 to encourage them to develop a modern series of airfoils specifically designed for General Aviation aircraft. Disappointed with NASA's near total lack of enthusiasm for such a project, he returned home, bought a computer, developed his own "GA" series of airfoils, and published his book, "GA Airfoils-a Catalog of Airfoils for General Aviation Use".

Harry and his wife Jo-Ann never miss going to the annual Oshkosh Convention, have three children (Allen, Gail and Mark), and six grand-children. Allen is an active "Starduster" pilot, and the others give "moral support".

June 1994

Harry Riblett
416 Riblett Lane
Wilmington, Delaware 19808
302/994-0479

ADDENDUM NO. 7

Aircraft Stability - Simplified

It is surprising how many builders ask this question - "I am using airfoil (XYZ) on my airplane. What should I use for the aft CG limit?" It would be nice if it were that simple, but it isn't. Aircraft longitudinal stability, and the aft CG limit, depend on many factors, and the particular airfoil being used is only one factor affecting the problem.

Case I - Flying Wing Airplanes. Let's begin by looking at the most simple case, a flying wing. As in any airplane, two conditions must be satisfied, that is, it must be statically stable, and it also must be dynamically stable. Static stability merely requires that the CG be located in line with the lift vector, that is, at the center of pressure of the airplane, so that the airplane (undisturbed) will fly level. Since we cannot shift the CG fore and aft in flight, we therefore need, for the flying wing airplane, an airfoil whose CP does not change position as the angle of attack changes, but remains at the "magic" quarter-chord point. This, by definition, is a "zero CM" airfoil, which means that the airfoil must be either symmetrical (uncambered), or, if camber is used in the airfoil section, the trailing edge must be reflexed sufficiently to achieve zero Cm. In practice, the trailing edge reflex can be changed slightly to trim out small changes in CG location, but only very small changes can be handled. We describe this "balanced" condition by saying that the sum of the moments about the CG must be zero, for any steady state fight condition.

Dynamic stability, on the other hand, involves the rate of change of the moments about the CG, following a disturbance from trimmed flight. To be dynamically stable, the disturbed airplane must somehow develop a net moment about the CG counter to the direction of the disturbance, that is, a net stabilizing moment, so that the airplane will tend to return to its trimmed attitude and speed. Notice that the flying wing airplane has precious little mechanism to develop this restoring moment. One possibility is to locate the CG beneath the wing as shown, so that when a nose up disturbance occurs, for example, the "pendulum effect" of the CG moving forward will tend to bring the nose back down to the the original position. Another more complicated possibility is to use electronic sensors to detect the direction and magnitude of the disturbance, so that appropriate actuation signals can be generated for the elevons, to produce the needed restoring moment about the CG.

From this we can see that flying wing airplanes involve some fairly serious design restrictions. First, zero pitching moment airfoils must be used, and these are comparatively inefficient in terms of L/D. Second, no flaps can be used on the airplane, for there is no tail to react the considerable nose-down pitching moment when the flaps are deployed. Accordingly, landing and take-off speeds are relatvely high, compared to a similar conventional airplane with flaps. Third, the CG range of the airplane is very limited, restricting the utility of the airplane. A flying wing airplane doesn't have to drag around a fuselage and a tail, but the price paid for this supposed advantage is too high.

Case II - Canard Airplanes. In a canard configuration, we have two wings operating in tandem. If we load the airplane per the attached sketch, with the CG 5ft forward of the aft wing C/4 point, we see that the forward wing produces 80 lbs of lift, and a positive moment of 800 ft-lbs about the CG. Similarly, the rear wing produces 160 lbs of lift, and a negative moment of 800 ft-lbs about the CG. Therefore, the total lift is 240 lbs, equaling the weight of the airplane, and the sum of the positive and negative moments about the CG is zero. Thus the resultant lift vector passes through the CG, and the airplane is statically stable.

If we now apply a 1 degree nose-up disturbance to the airplane, for example with a sudden stick movement, or with a nose-up gust load, the picture changes. Since the slope of the lift curve for any airfoil is approximately 0.1 change of Cl for each one-degree change of angle of attack, the Cl of the forward wing will increase from Cl=.4 to Cl=.5, which is a lift increase of only 25%, while the Cl of the rear wing will increase from Cl=.2 to Cl=.3, a lift increase of 50%. This is the key to achieving dynamic stability. Completing the analysis, the lift of the forward wing increases from 8Q to 10Q (you're welcome), and the lift of the rear wing increases from 16Q to to 24Q, for a total resultant lift of 34Q lbs. But since the weight of the airplane is still only 24Q lbs, the first thing we notice is that the airplane goes up. Now, taking moments about the CG, we see that the resultant CP is no longer aligned with the CG, which first of all means that the airplane is in a transitional condition. The net moment on the airplane about the CG is now negative (nose down), and amounts to (34Q lbs)(0.6 ft)=20.4Q ft-lbs. This is the stabilizing moment that we are looking for, to bring the nose back down toward the trimmed attitude.

There are other lesser factors that affect dynamic stability, such as forward fuselage area, friction in the controls, and stick-free (destabilizing), and aft fuselage area and stick-fixed (stabilizing), as well as propeller and downwash effects. These are not included in the above simple analysis, but the principles remain the same. As long as the CG is forward of the airplane's neutral point, a net stabilizing moment will result after a disturbance. In our analysis, neglecting the lesser factors mentioned above, the neutral point is the CG location at which the Cl of both wings is the same.

From the above we can see that the primary requirement for any tandem wing airplane regarding dynamic stability is that the forward wing must be more highly loaded than the rear wing. In addition, the forward wing must always stall before the rear wing, so that the nose of the airplane will drop at the stall, rather than the reverse. In the case of the canard airplane, this means that the main wing can never reach its maximum lift coefficient, which spells inefficiency and relatively high landing speeds. Further, it means that flaps cannot be used on the main wings of a canard airplane. A further restriction is that a canard airplane should not be flared on landing, since a stall of the forward wing approaching touch-down is likely to cause a sudden nose drop.

Case III - Conventional Airplane. A conventional airplane (tail in the rear) is also a tandem wing airplane, so it obeys the same rules for stability as stated above. The horizontal tail surface, however, unlike a canard, is called on to produce a lift in either direction, up or down. Accordingly, the usual practice is to compromise, and use a symmetrical section for the tail surfaces, such as the NACA 63A012, which has a good L/D ratio and decent thickness. By using a relatively large, well designed tail, we can achieve a wide CG range. This is a desirable design objective, for instance in a cabin-class airplane, enhancing the utility of the airplane by permitting considerable flexibility in loading. By way of illustration, let's examine an airplane with a main wing area of 80 sqft, and a horizontal tail area of 20 sqft, or 25% of the main wing area. This is slightly more than the rule of thumb of 15-20% of the wing area, used on smaller airplanes. Now suppose our airplane uses a wing section with a pitching moment of Cm=-0.04, and suppose the design lift coefficient at cruise is Cldes=0.2. Referring to the attached CP chart, we see that the main wing CP is at .45C at cruise. If we load the airplane with the CG at this same point, .45C (aft CG limit), the airplane will be statically stable, and the lift coefficient of the tail will be zero. Further, we see that this is also a dynamically stable condition, since the lift coefficient of the rear wing (tail) is lower than the lift coefficient of theforward wing. We also notice that this is a particularly efficient configuration, since there is no wasteful tail down-load, which would occur if the CG were placed farther forward, or if the pitching moment coefficient of the wing were greater, or both. For example, if the airplane used the NASA GAW-1 airfoil, with its outrageous pitching moment coefficient of Cm=-0.12, the CP of the main wing would be off the chart, and a considerable (and wasteful) tail down-load would be required to trim the airplane.

Next we check to make sure the tail does not stall before the main wing. At the stall, our wing airfoil has a Clmax of 1.8, and the CP of the main wing moves forward at or very close to the C/4 position, as all airfoils do. With the CG at .45C, we see that the required tail lift coefficient will be +0.51 for static stability. This is within the range of our NACA 63A012 tail section, even at the relatively low Reynolds number and low aspect ratio common for tail surfaces, and we are satisfied that the tail will not stall before the main wing. We also note that the tail Cl is considerably less than the wing Cl, so this condition is dynamically stable as well. We probably wouldn't have been able to prove out this aft CG limit with a smaller, poorly designed tail, such as the flat tails of tubular construction common to light airplanes.

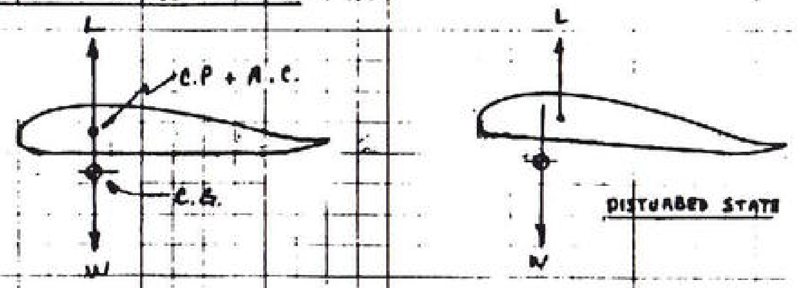
We are not finished yet, however. Before we can be satisfied that this is a safe aft CG limit, we must also investigate the spin recovery properties of the airplane at this CG location. This involves a whole new set of parameters, beyond the scope of this paper.

It is interesting to note, as shown by the example above, that a small amount of pitching moment coefficient in an airfoil can actually be helpful in achieving a wider CG range. Thus the zero-Cm airfoils that NACA developed in the 5-digit studies serve no useful purpose, at least in conventional airplanes. At the other extreme, the later NASA GAW-1 and -2 airfoils, which have Cm values 3 times larger than they need to be, are equally ill-advised.

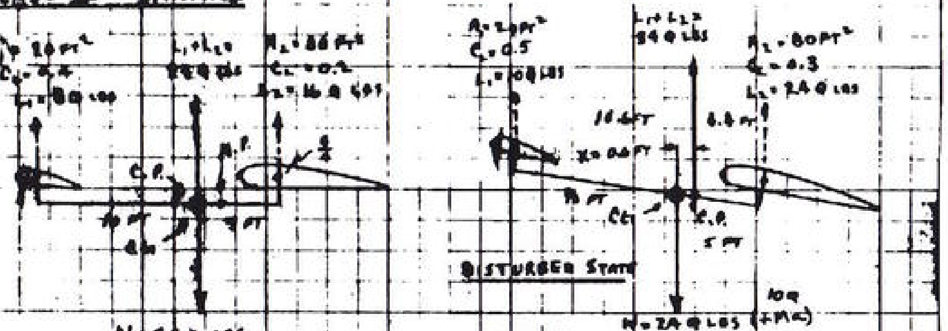
How shall we answer the question posed in the first paragraph above? The correct answer is "The aft CG limit should be slightly forward of the aircraft neutral point, provided the tail does not stall before the main wing, and provided the spin recovery characteristics of the airplane are satisfactory." However, this is difficult to calculate prior to flight tests, therefore a conservative practice with simple airplanes is to ignore the influence of the tail on the position of the airplane's neutral point, and simply put the aft CG limit at or near the wing A.C., that is, at approximately 25-30% of chord. This seems to be a safe practice, at least until flight tests can be done to possibly extend the CG range. Remember, however, that the CG pendulum effect works for us on high wing airplanes, but works against us on low wing airplanes. Thus, for low wing airplanes, 23-25%C is often used as the aft CG limit prior to flight tests.

Harry Riblett
4/11/1996

CASE I - FLYING WING


L
C.P + A.C.
C.G.
W
L
W
DISTURBED STATE

CASE II - CANARD


A₁ = 20 FT²
C_L = 0.4
L₁ = 80 LBS
L₁+L₂ =
240 LBS
A₂ = 80 FT²
C_L = 0.2
L₂ = 160 LBS
C.P.
A.P.
c/4
10 FT
5 FT
CG
W = 240 LBS
A₁ = 20 FT²
C_L = 0.5
L₁ = 100 LBS
L₁+L₂
340 LBS
A₂ = 80 FT²
C_L = 0.3
L₂ = 240 LBS
10 FT
6.6 FT
X = 0.6 FT
10 FT
CG
C.P.
5 FT
DISTURBED STATE
10#
W = 240 LBS (+Ma)

$[\Sigma M_{cg}=0]$

$80^{\#}(10\,FT) = 160^{\#}(5\,FT)$ ✓

$[\Sigma M_{cg}=0]$

$(240)(5\,FT) - (100)(10\,FT) = 340(x\,FT)$

$x = 120-100/34 = \underline{0.6\,FT}$ ✓

CASE III - CONVENTIONAL (SPECIAL CASE - 0 TAIL LOAD)

$A_w = 80\,FT^2$

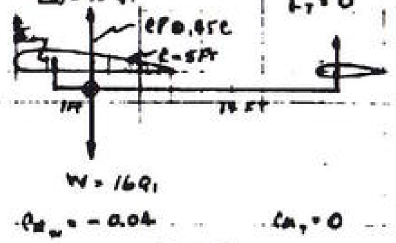
$C_{L_W} = 0.2$

$L_w = 16\,q_1$

$A_T = 20\,FT^2$

$C_{L_T} = 0$

$L_T = 0$


CP@.45c
C = 5FT
1FT
14 FT

$W = 16q_1$

$C_{M_W} = -0.04$ $C_{M_T} = 0$

CRUISE

$A_w = 80\,FT^2$

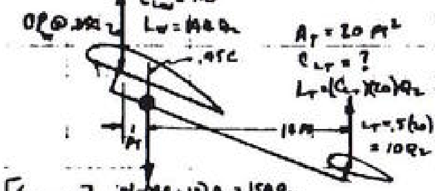
$C_{L_W} = 1.8$

$L_w = 144\,q_2$

$A_T = 20\,FT^2$

$C_{L_T} = ?$

$L_T = (C_{L_T})(20)q_2$

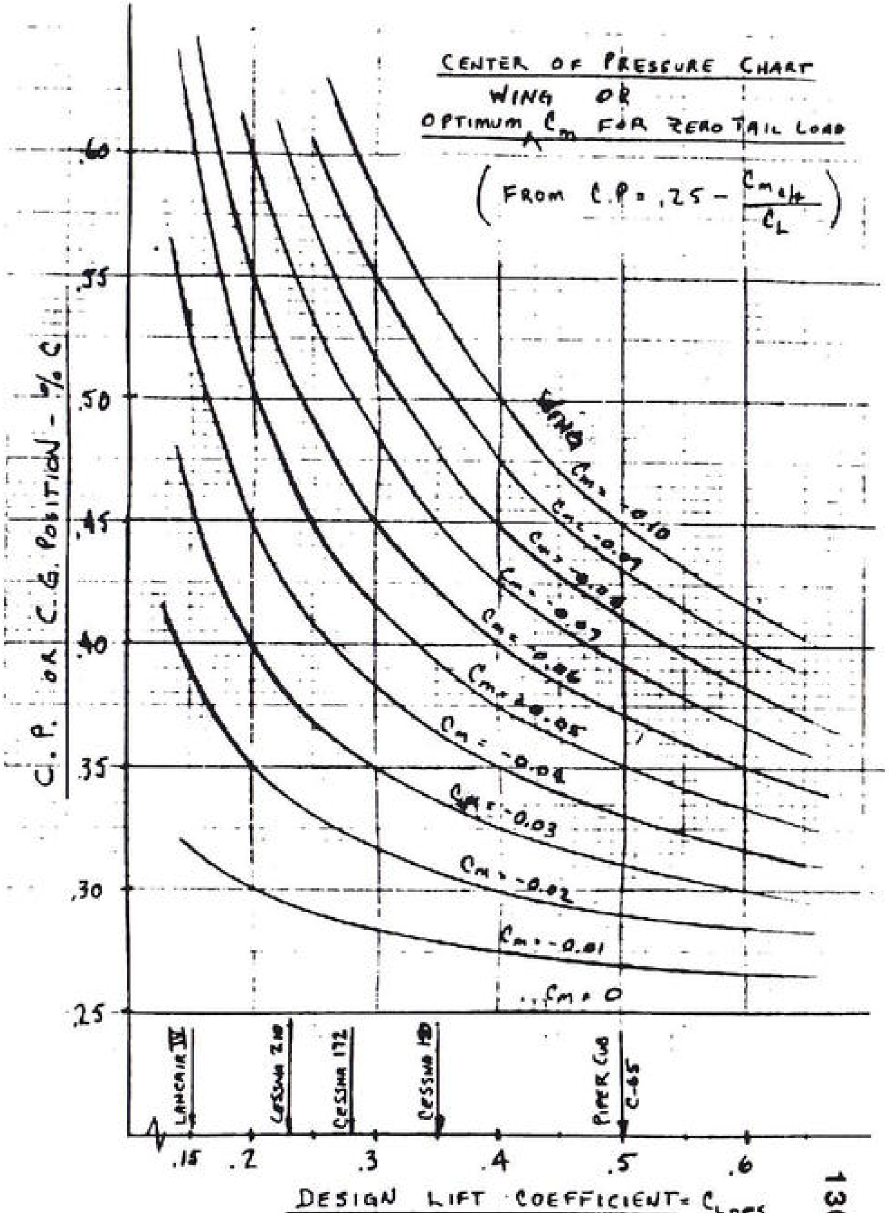

CP@ [illegible]
.45c
1FT
14 FT
L_T = .5(20)
= 10q₂

$[\Sigma M_{cg}=0]$ $W = (144+10)q_2 = 154q_2$

$(1\,FT)144q_2 = (C_{L_T})(20\,FT^2)(14\,FT)q_2$

$C_{L_T} = \frac{144q_2}{(20)(14)q_2} = \underline{0.51}$ OK ✓

STALL - NO FLAPS



CENTER OF PRESSURE CHART
WING OR
OPTIMUM C_m FOR ZERO TAIL LOAD

$\left(\text{FROM } C.P. = .25 - \frac{C_{m_{c/4}}}{C_L}\right)$

DESIGN LIFT COEFFICIENT = $C_{L_{DES}}$

HR - 3/26/96

ADDENDUM NUMBER 8

Vortex Generator Kits

This paper discusses the highly effective vortex generator kits that are available to lower stall speeds and improve low-speed controllability, especially on prop-driven twins with one engine out. The subject was discussed at some length in my papers "Design Notes for Tapered Wings", and "Taming the Sharp Stall of the NACA 23012 Airfoil". We also elaborate on an article concerning Micro-Dynamics Inc VG kits printed in the January 1996 issue of "Air Progress" magazine, and we strongly recommend that anyone interested in the subject should read that excellent article.

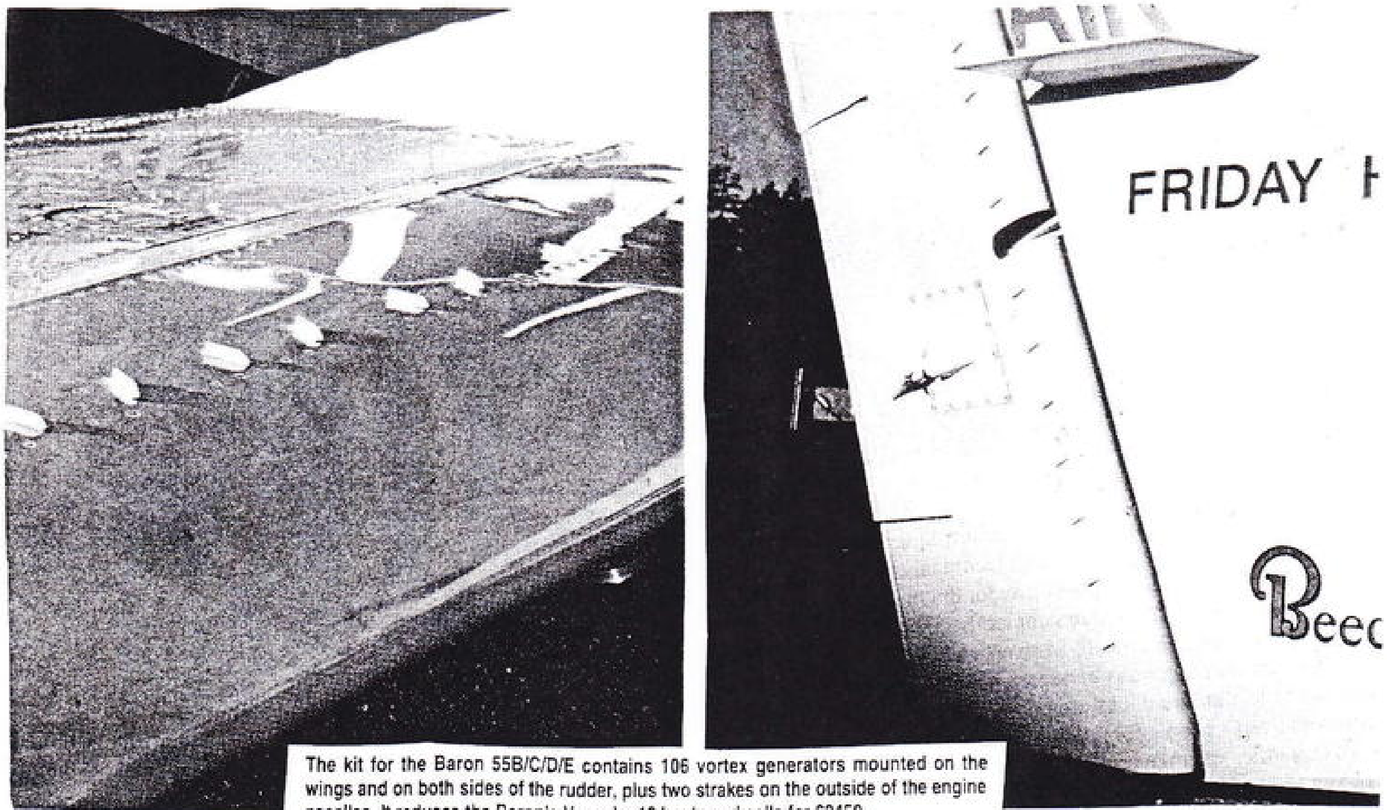
The VG kits basically contain two sets of vortex generators. One set is installed on the top surface of the wing, across the entire wingspan, at the 10% chord position. The second array is installed as close as possible to the leading edge of the rudder, both sides. In some cases, this second set is installed near the trailing edge of the fin. The primary effect of the VG's is to increase the thickness of the boundary layer downstream of the VG's. Thus, they increase the effective thickness of the wing, just as if "Bondo" were applied to the top surface. And since the added thickness is all on the top surface of the wing, this causes an upward shift of the airfoil mean line, that is, an increase of wing camber. Both the increased thickness and the increased camber result in a higher maximum lift coefficient for the wing. More importantly, however, these VG's, when placed at the .10C position on the NACA 2301x airfoils that are used on most GA light twins, eliminate or at least soften the needless and troublesome discontinuity in the mean line that these airfoils have at the .15C position. This eliminates the flow separation that occurs at that point at high angles of attack. Without the flow separation, the stall becomes soft and gentle, and this is the key to good aileron response, even with one engine out. Thus, for the 2301x airfoils, the VG's give improved controllability (elimination of Vmc) as well as a higher Clmax, whereas on the Cubs and Champs, which already have soft-stall airfoils, the primary effect is simply an increase in Clmax.

The rudder VG's reduce flow separation on the low pressure side of the deflected rudder. Without VG's, the simple hinged rudders used on most of today's light twins have terrible flow separation at the hinge line, thus high drag, when one engine quits and hard rudder opposite the dead engine is applied to hold heading. In this condition, the rudder becomes a large, effective drag brake, just what we don't need.

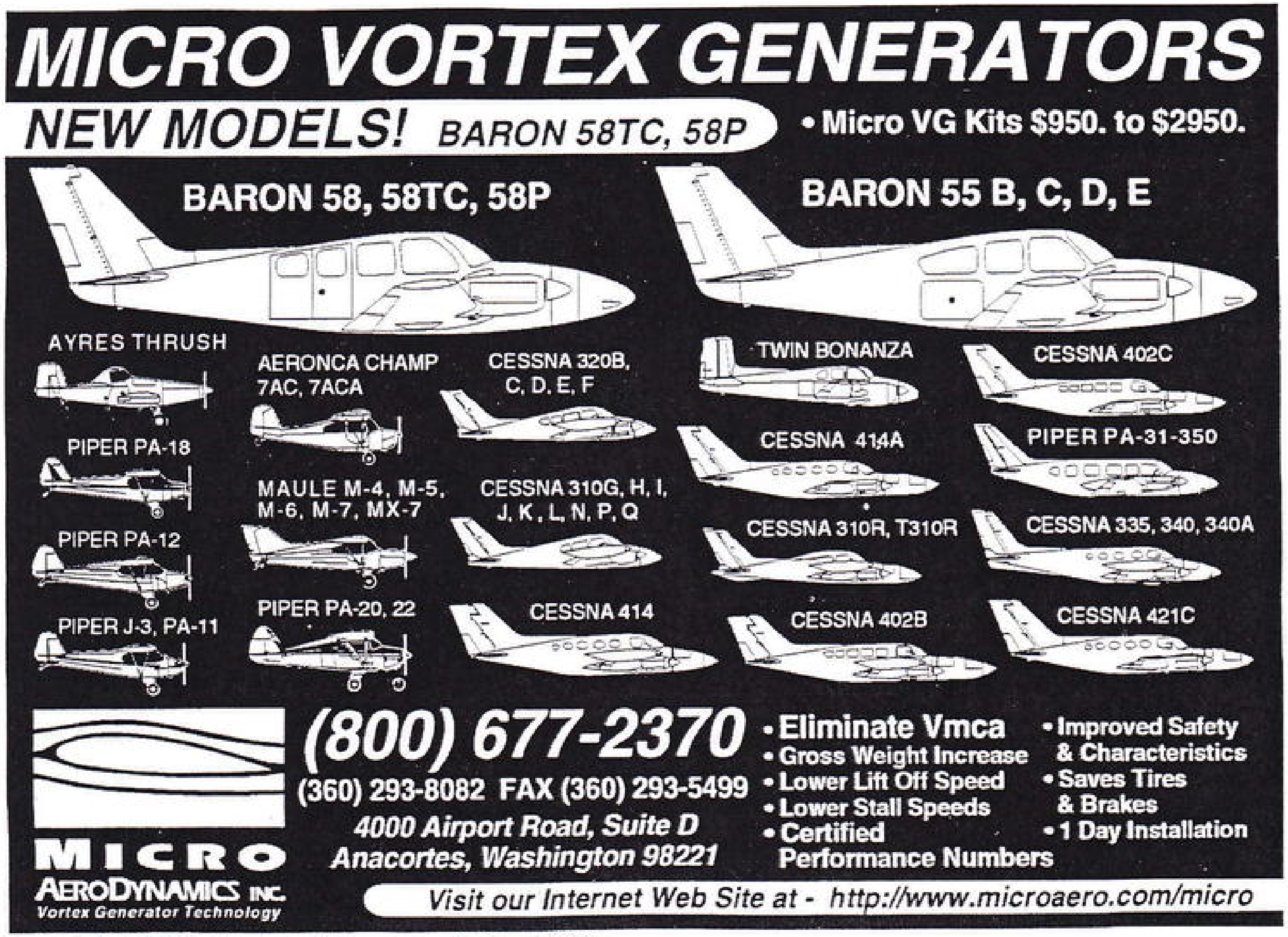
Thus, the VG's are very effective band-aids that address the poor wing and tail designs that are used on today's light twins (and commuter twins as well). Of course, the better approach is to design and build the planes properly in the first place. Three things are required, all within current technology: (1) Soft stall airfoils, such as "GA" airfoils (2) Wing tips at least 12% thick for turbulent airfoils, and 15% thick for laminar flow airfoils, and (3) Low drag, effective rudder designs, such as the articulated rudder used on the USAF C-17.

Why does Wichita continue to build this trash, when we know better? It's the fault of the FAA type certificate system, for the re-certification costs associated with major design changes effectively freezes the design. Thus we are forever wedded to 40 and 50 year-old technology, to the detriment of safety. This must be changed. The first thing that FAA should do is to rescind (after a suitable grace period of perhaps two years) the type certificates of all prop-driven twins that cannot show a Vmc lower than the normal stall speed of the airplane. Then the amended type certificate for the improved "no Vmc" model should be issued following a simple flight demonstration, without the prohibitively expensive "normal" certification procedure. Imported airplanes would have to meet the same performance requirements. With that, we would at last have true twin-engine reliability, which we do not have with today's obsolete, garbage airplanes.

Harry Riblett - 5/17/1996



The kit for the Baron 55B/C/D/E contains 106 vortex generators mounted on the wings and on both sides of the rudder, plus two strakes on the outside of the engine nacelles. It reduces the Baron's Vmca by 10 knots and sells for $2450.



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CESSNA 414
TWIN BONANZA
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CESSNA 402C
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FIGURE 1